CONTENTS

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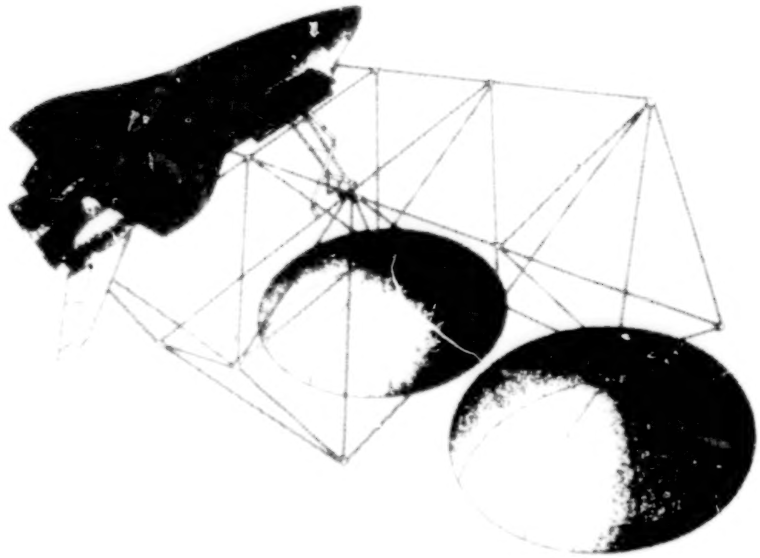
Large Space Systems Technology



An Industry/Government Seminar
held at NASA Langley Research Center
Hampton, Virginia, January 17-19, 1978

NASA

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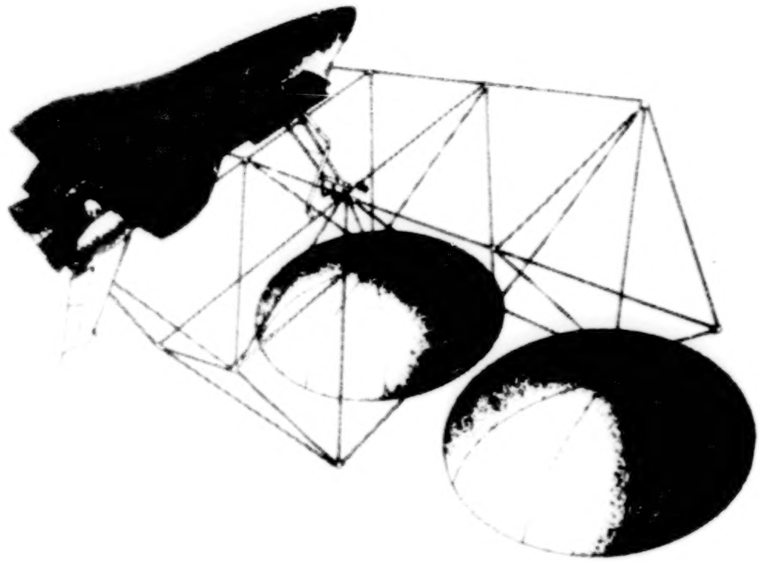

Large Space Systems Technology

Compiled by
E. C. Naumann
Langley Research Center
and
A. Butterfield
General Electric Company

An Industry/Government Seminar
held at NASA Langley Research Center
Hampton, Virginia, January 17-19, 1978

National Aeronautics
and Space Administration

Scientific and Technical
Information Office

1978

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PREFACE

These Proceedings include a compilation of the papers presented during the Seminar and a summarization of pertinent questions and answers arising during the presentations. The Seminar provided Government and Industry representatives with an opportunity to exchange information, to review the present status of related technology, and to plan the development of new technology for Large Space Systems. The Seminar began with four invited papers: three NASA papers to provide industry and the user community with information pertinent to NASA's planning and forecasting efforts, and one Industry paper to solicit for NASA, by way of a questionnaire to industry, information that would help NASA in planning the total program.

Selected papers, solicited from Government and industry, describing items of technology or developmental efforts followed the invited papers. These papers were divided generally into two major areas of interest: the first group addressed subjects pertinent to large antenna systems; the second group addressed technology related to large space platform systems. The Seminar concluded with a Forum and Issues session with participants from both Industry and Government.

This compilation provides the participants and their organizations, in a referenceable format, the papers presented at the Seminar. The Large Space Systems Technology Program Office, Langley Research Center, which sponsored the Seminar, will utilize this information as an aid in their planning and in the definition of technology goals and technology developments.

At NASA LaRC, the work was monitored by E. C. Naumann, Task Manager and Seminar Coordinator. A. Guastaferro, Manager of the LSST Program Office served as General Chairman of the Seminar. The Task was administered by S. M. Scala, Senior Consulting Scientist, General Electric Company, who also served as Moderator of the Forum/Issues Panel.

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ACRONYMS AND ABBREVIATIONS

The list of acronyms and abbreviations contains both commonly used terms and specific items not immediately defined within a presentation. Single usage acronyms readily defined within the text, U.S. Government agencies, and spacecraft acronyms do not appear. In addition, the technology for Large Space Systems has begun to generate descriptive words coined or extracted from common usage terms; such words with obvious meaning have not been included.

ACS	Altitude Control System
AMB	Amplitude Modulated Beam
AMVSB	Amplitude Modulated Vestigial Side Band
ATS -(6)	Advanced Technology Satellites (No.)
AWG	American Wire Gage
BER	Bit Error Rate
BW	Bandwidth or Beamwidth
CCD	Charge Coupled Devices
CCIR	International Radio Consultative Committee
Cervet	Trade Name for a Glass Ceramic with a very low Coefficient of Thermal Expansion
C.G.	Center of Gravity
CMG	Control Moment Gyro
CMD	Command
CONUS	Continental United States
CoTE	Coefficient of Thermal Expansion
CRT	Cathode Ray Tube Display
CW	Continuous Wave

Cy, CY	Calendar Year (Jan. 1 thru Dec. 31, inc.)
DCP	Data Collection Platforms
DDR&E	Department of Defense, Research and Engineering
DOF	Degrees of Freedom
DSN	Deep Space Network
d/t	Ratio of Diameter to Wall Thickness for a Tube
EM	Electro-Magnetic
ERP	Effective Radiated Power
E	Extensional stiffness
EVA	Extra Vehicular Activity
f, F	Frequency
FUV	Far Ultra Violet Light
Fy, FY	Fiscal Year (now Oct. 1 through Sept. 30, inclusive)
g, G	Gravity
GCP	Geostationary Communication Platform
GEO	Geosynchronous Earth Orbit
GG	Gravity Gradient
GPC	General Purpose Computer
GRC	General Research Corp.
GR/E	Graphite Epoxy
(GY--)	(Graphite Epoxy Formulation Code)

GTD	Geometric Theory of Diffraction
GTP (GR/TP)	Graphite Thermoplastic
h	Film Transfer Coefficient for Thermal Conductivity
HEO	High Earth Orbit
HMS	Graphite Fiber Trade Mark, High Modulus Series
IC	Integrated Circuit (Microcircuit)
ICD	Interface Control Document
IF	Intermediate Frequency
INC	International Nickel Co.
IR	Infra Red
IR&D	Independent Research and Development
IRU	Inertial Reference Unit
ISP	Specific Impulse
IUS	Interim Upper Stage
L	Altitude in Equivalent Earth Radii
LED	Light Emitting Diode
LEO	Low Earth Orbit
LEPS	Laser Electric Propulsion System
LSS	Large Space Systems
LSST (LASST)	Large (Area) Space System Technology
(ATLASS)	(Advanced Technology for LASS)

LTL	Low Thrust Liquids
MCDS	Manual Control Display System
MCDU	Manual Control Interface Unit
MDM	Multiplexer-Demultiplexer
MF	Mass Fraction (Percent of total weight carried as fuel)
MFG	Manufactured
MHD	Magnet Hydro Dynamic
MMH	Mono methyl Hydrazine
MOUSE	Computer Program for Modal Optimization and Analytical Model Updating (Test data)
NASCAP	Computer Program for Analyses of Electrical Charges
Nd/YAG	Neodymium doped Yttrium Aluminum Garnet
NE ΔT	Noise Equivalent Temperature
NF	No Failure
NUV	Near Ultra Violet
OFT	Orbital Flight Test
OMS	Orbital Maneuvering System
OOA	On-Orbit Assembly
OTV	Orbit Transfer Vehicle
P	Protons

P, $Pcr_{(exp, th)}$	Compression Load applied to a column, cr, loading which causes buckling (experimental, theoretical)
Paramp	Parametric Amplifier
POP	Program Operating Plan
RBV	Return Beam Vidicon
R&D	Research and Development
R&T	Research and Technology
RDT & E	Research Development Test and Engineering
RCS	Reaction Control System
rf RF	Radio frequency
Re	Earth Radius
RFP	Request for Proposal
RMS, rms	Root Mean Square for surface accuracy or statistical error
RMS	Remote Manipulator System
RT	Room Temperature
RTG	Radio isotope Thermoelectric Generator
RTOP	Research and Technology Objectives and Plans
RTR	Research Technology Resume
RTV	Trade name for silicone elastomeric materials
SAR	Synthetic Aperture Radar
SEPS	Solar Electric Propulsion Stage
SL	Side Lobe

S/N	Signal to Noise Ratio
SPAR	SPAR Aerospace Products Ltd.
SPAR	Computer Program for General Purpose Structural Analysis
SPS (SSPS)	Solar Power Satellite (Space SPS)
STS	Space Transportation System
TASO	Television Allocation Study Organization
TDRSS	Tracking and Data Relay Satellite System
Tg	Glass Transition Temperature
TR-SW	Transmit-Receive Switch
TV	Television
TVBS	Television Broadcast Satellite
T/W	Thrust to Weight Ratio
UV	Ultra Violet
UDT --	Electronic Part Numbers for United Technology Devices
ULE	Trade name for a Glass Ceramic with a very low coefficient of Thermal Expansion
VCM	Vapor Condensable Material
VSB	Graphite Fiber Trademark

ABBREVIATIONS FOR UNITS AND MEASUREMENTS

The following symbols and abbreviations have been used within this compilation.

Å	Angstrom units, 10^{-10} meter
A, amp	amperes
AU	Astronomical Unit
arc min	angular measurement in minutes of a degree
arc sec	angular measurement in seconds of a degree
b/sec, B/SEC	bits per second
BTU	British thermal unit
C, °C	Celsius degrees
cm, CM	centimeters
db, dB, DB	decibels
dbw, DBW	decibels relative to one watt
eV, EV	electron volts
F, °F	Fahrenheit degrees
ft., FT,	feet
G -	giga (X 10^9) multiplier
GHz	gigahertz
GW	gigawatt
GWe	gigawatt electrical

gm, Gm	grams
g/cm^2	area density, grams per square centimeter
Hz	hertz
in, IN, "	inches
°K, K	degrees Kelvin
k-K	kilo ($X\ 10^3$) multiplier
KA	kiloampere
Kb/s, KB/S KBPS	kilobits per second
kev, KEV	kilo electron volts
kg, KG	kilogram
km, KM	kilometer
KMC	kilomegacycles (same as GHz)
kv, KV	kilovolt
kw, KW	kilowatt
kw-hr	kilowatt hours
Kwe, KWE	kilowatt electrical
lbs, LBS	pounds
lb-ft	pound feet (torque)
m, M	meters

M -	mega, million (10^6) multiplier
M bits/sec	megabits per second
MHz	megahertz
MW	megawatt
MWe	megawatt electrical
meV, MEV	million electron volts
Micron	10^{-6} meter
MI	mile
Mil	10^{-3} inch
m-- M--	Milli -- (10^{-3}) multiplier
mm, MM	Millimeter 10^{-3} meters
ms	milliseconds
nm, NM	nautical miles
N-m, N-M	Newton meters (torque)
n-m-s, N-M-S	angular momentum, Newton-meters-seconds
psi, PSI	pounds per square inch
rad, RAD	radian
RAD/SEC	radians of angle per second
ST MI	Statute mile
TORR	pressure, equivalent to millimeters of mercury
V	Volts
W	watts

λ	wavelength
μ	micro (10^{-6}) multiplier
μm	10^{-6} meter
$\downarrow + \uparrow$	perform the operation in two opposing orientations relative to the force of gravity
#	Number of

CONVERSION FACTORS FOR UNITS

U.S. Customary Units	SI Units	Multiply by
ampere (International)	ampere	0.9998
BTU	joule	1055
electron volt	joule	1.6×10^{-19}
Fahrenheit (temperature)	Kelvin	$5/9\ (t_F + 459.67)$
Fahrenheit (temperature)	Celsius	$5/9\ (T_f - 32)$
foot	meter	0.3048
inch	meter	0.0254
pound force	newton	4.448
pound mass	kilogram	0.4536
mile	meter	1.609×10^3
nautical mile	meter	1.852×10^3
slug	kilogram	14.594
psi	pascals	6.895×10^3

INTRODUCTION

The availability of the Space Shuttle transportation system will make it possible to deploy, erect and/or eventually fabricate on orbit large space systems (LSS) beginning in the decade of the 1980's. Preliminary studies conducted by NASA, DOD, and the Aerospace Industry indicate that in order to meet future user needs, large antennas and platforms will be required either in low Earth orbit or in geo-synchronous orbit. Specific applications have been identified in a series of recent studies which have examined and evaluated future civilian and military space possibilities with relevance to human and/or defense needs as the basic measures.

One of the most comprehensive recent studies resulted in two NASA reports, issued in January 1976: "Outlook for Space" (NASA SP-386), which identified potential future space activities, and "A Forecast of Space Technology" (NASA SP-387) which provided a comprehensive forecast of technology which might reasonably be expected to be available for the management of information, energy, and matter in space during the last two decades of the 20th century.

A three-day Industry Workshop on Large Space Structures was held at the NASA Langley Research Center in February 1976, to help NASA identify the technology developments required for these proposed missions. At this workshop, representatives of several Aerospace Companies were asked to respond to a Key Issue Questionnaire. These responses were published in two NASA reports: A Compilation of Company Presentations (NASA CR-144997); and An Executive Summary (NASA CR-2709).

In March 1977, the Langley Research Center was named lead Center of a multi-center, multidisciplined planning activity with the mission of defining and developing critical technology for use in large space systems in the years 1985 to 2000. The Large Space Systems Technology (LSST) Program Office evolved from these planning and program definition activities. This seminar was sponsored by the LSST Program Office to provide a forum for the more effective interchange of ideas, plans, and program information needed to develop the required large space system technology. The format of the Seminar is closely aligned to that of the 1976 workshop because of the effective interchange obtained during the workshop.

The seminar organizing committee utilized invited papers, contributed papers, and panel discussions to maximize potential benefits for each of the participating organizations. The invited papers were used to provide industry an insight on the views of NASA Headquarters, the LSST Program Office, and background information on Shuttle astronaut interfaces, and to provide NASA information on Industry views by means of a questionnaire. The contributed presentations were essentially equally divided between industry and Government. These papers emphasized on-going or planned in-house technology development in support of large antenna systems or large platform systems. Typical subject matter

addressed at least one of the following: mission requirements, structural concepts, materials, controls, structural alignment, thermal control, metrology, and packaging/shuttle interface considerations. Finally, the last session of the Seminar was devoted to a Forum, the purpose of which was to provide industry and government representatives with the opportunity to present their views on significant and/or controversial issues, to answer questions from the attendees, and to focus attention on critical LSST needs and approaches.

The proceedings of the Seminar are presented in this report and in an Executive Summary (NASA CR-2964). This report contains all the invited and contributed formal presentations given during the Seminar and also includes those contributed papers which were considered to be informative and useful but which could not be delivered during the program due to time constraints. The Executive Summary condenses and records the findings and conclusions of all the presentations and, in addition, highlights the comments and recommendations of the members of the Forum Panel.

OVERVIEW OF THE LARGE SPACE SYSTEMS TECHNOLOGY PROGRAM

A. GUASTAFERRO

LSST PROGRAM OFFICE

LANGLEY RESEARCH CENTER

GOVERNMENT / INDUSTRY SEMINAR

JANUARY 17 - 19, 1978

TECHNOLOGY FOR LARGE SPACE SYSTEMS (Figure 1)

Objectives

The broad objective of the Large Space Systems Technology (LSST) Program is to define and develop the necessary technology for large space systems and associated subsystems required for projected NASA space missions in the 1985-2000 time period. It is a goal of LSST to make these systems economically as well as technically feasible by focusing on those technical activities believed to provide the greatest benefit to a variety of future systems.

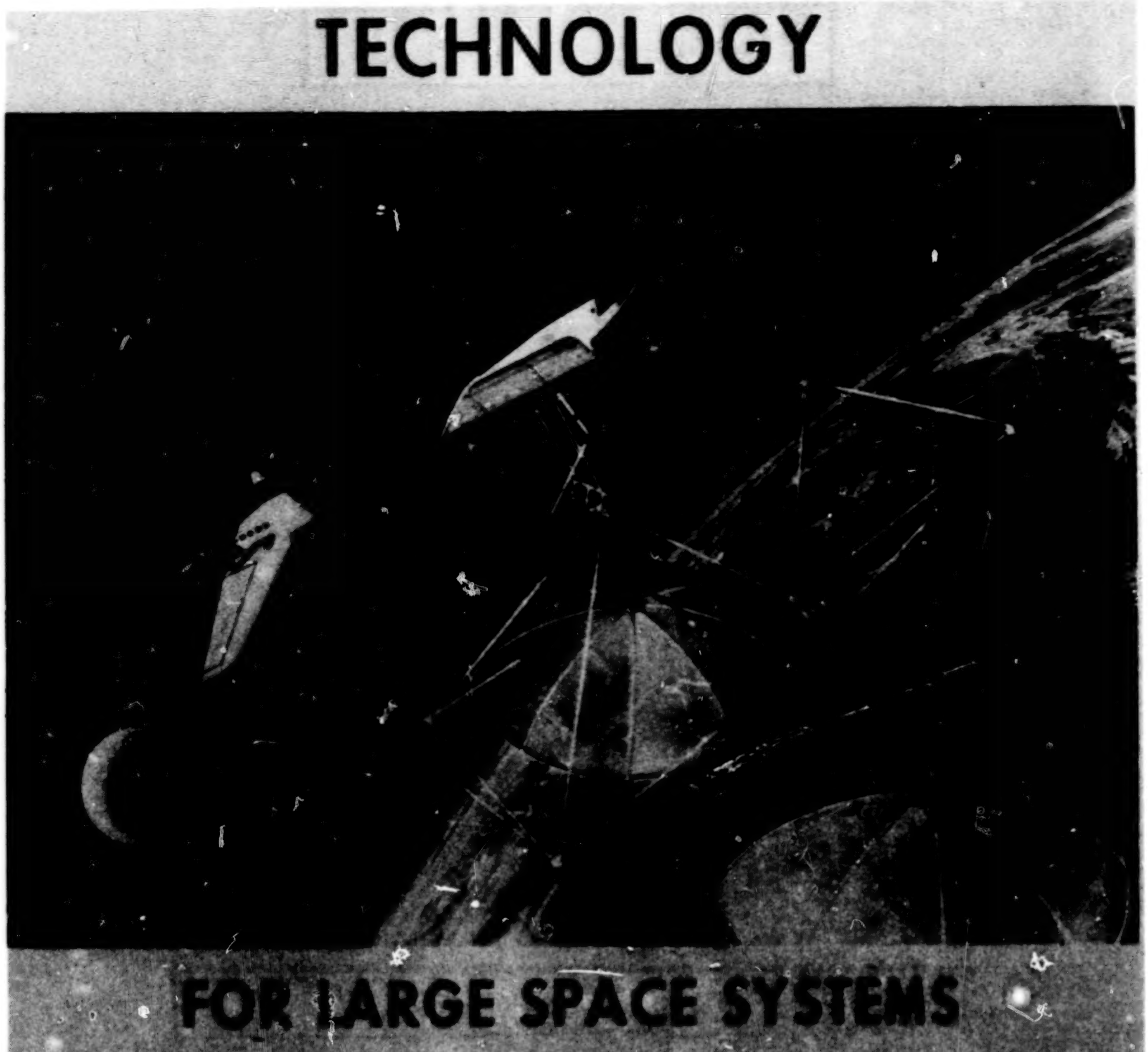


Figure 1

WHY LSST? (Figure 2)

As the United States shifts from Shuttle development to Shuttle operations, it is obvious that space systems of the future can take on new dimensions in size, complexity, power requirements, etc.

Several of the missions proposed for the 1985-2000 time period will result in cost and ecological benefits to the nation. Many require large space systems and associated subsystems beyond the state-of-the-art of today's technology. It is a goal of LSST to make these systems economically as well as technically feasible by focusing on those technical areas which will provide the greatest benefits to a variety of future space systems.

LSST technology will reduce design and development costs for future large space systems by providing developed and verified structural concepts, analyses, and design procedures for a range of sizes and configurations. LSST will reduce Shuttle transportation costs by developing concepts which have relatively low masses, high packageability and multi-mission capabilities.

An underlying program objective is to contribute to National progress, not only by enhancing space systems design technology, but also by identifying LSST-produced technologies which can assist in the solution of problems in other sectors of the national economy. These identified technologies will be made available to the Technology Utilization Program for dissemination.

The LSST Program approach to provide the necessary integrated technology for a variety of users of large space systems is somewhat new and has many potential benefits, including the broadening of narrow, individual mission requirements into a broad matrix of requirements.

WHY LSST?

- Operational Shuttle will provide the Nation with the opportunity for future space systems that require
  - more reliable predictive capability
  - significantly larger structures
  - more complex control systems
  - deployment, assembly and/or fabrication capability "On Orbit"
  - integrated design of structure/electronics/power
  - greater surface accuracy
  - longer operational life times
  - greater disciplinary interaction

- LSST will develop during the FY 79 to FY 84 time period
  - a systematic method of evaluating technology requirements based on;
    - Needs
    - Technology Gaps
    - Mission Commonality
  - an evaluation method to be used to establish program [illegible] and priorities
  - the technologies needed to achieve economically and [illegible] concepts for [illegible] of missions by the 1985-[illegible]

LSST

Figure [illegible]

NASA [illegible]-77-[illegible]

LARGE SPACE SYSTEMS TECHNOLOGY PROGRAM (Figure 3)

The LSST Program is managed by the Langley Research Center (LaRC) as lead center. The Program Manager reports to the LSST Office in the Materials and Structures Division of OAST. The LaRC-LSST Program Office (LSSTPO) is supported by four NASA Centers and JPL through their designated representatives. Each of these center representatives reports to the Program Manager who is responsible for integrating the efforts of the five Centers into a program that is responsive to the needs of large space systems technology developments.

The LSST approach is keyed to a centrally managed, multi-center management approach which provides the opportunity to work across the multi-disciplines and to match the roles and missions (expertise) of NASA Centers.

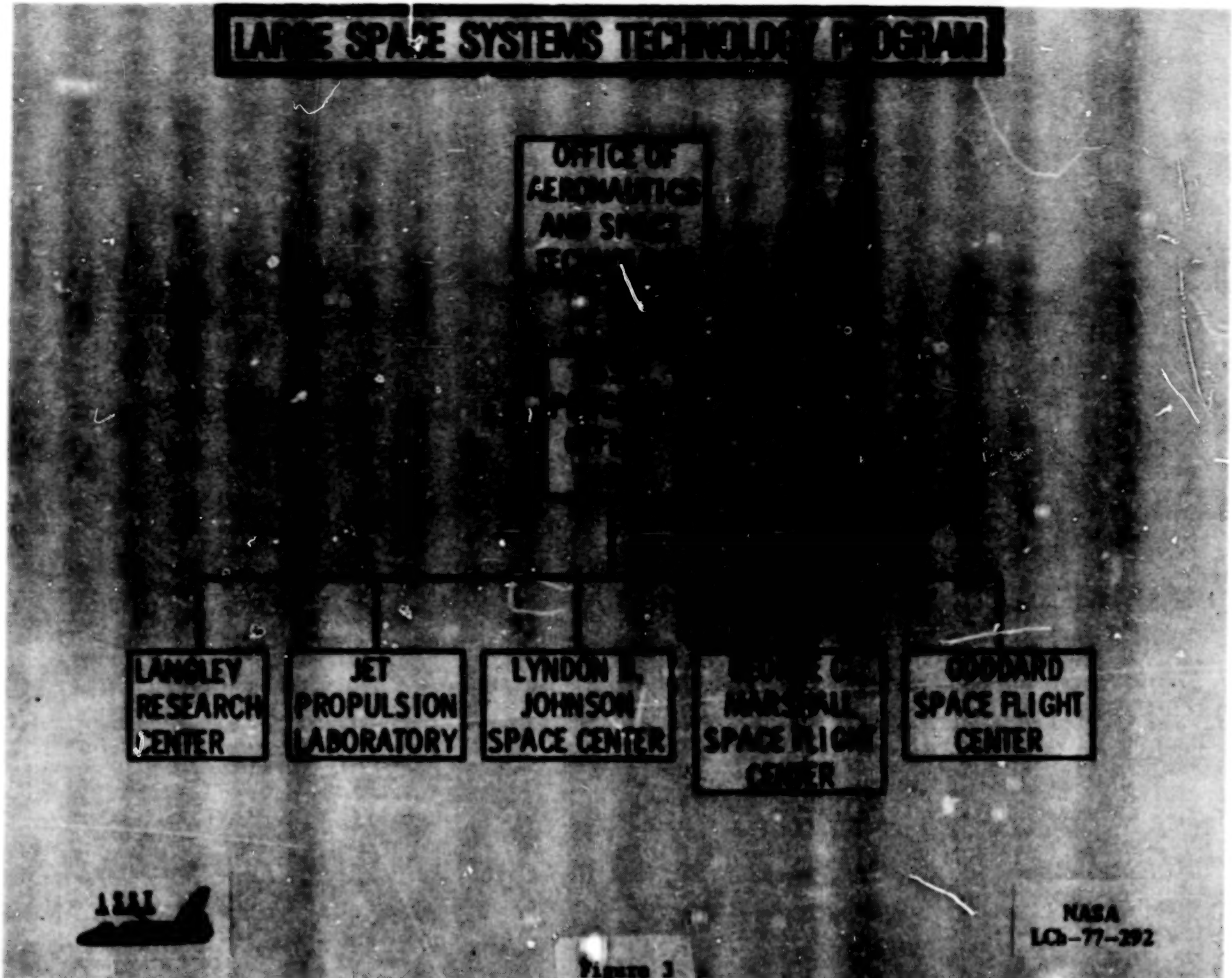

LARGE SPACE SYSTEMS TECHNOLOGY PROGRAM
OFFICE OF AERONAUTICS
LANGLEY RESEARCH CENTER
JET PROPULSION LABORATORY
JOHNSON SPACE CENTER
GODDARD SPACE FLIGHT CENTER
NASA
L-77-292

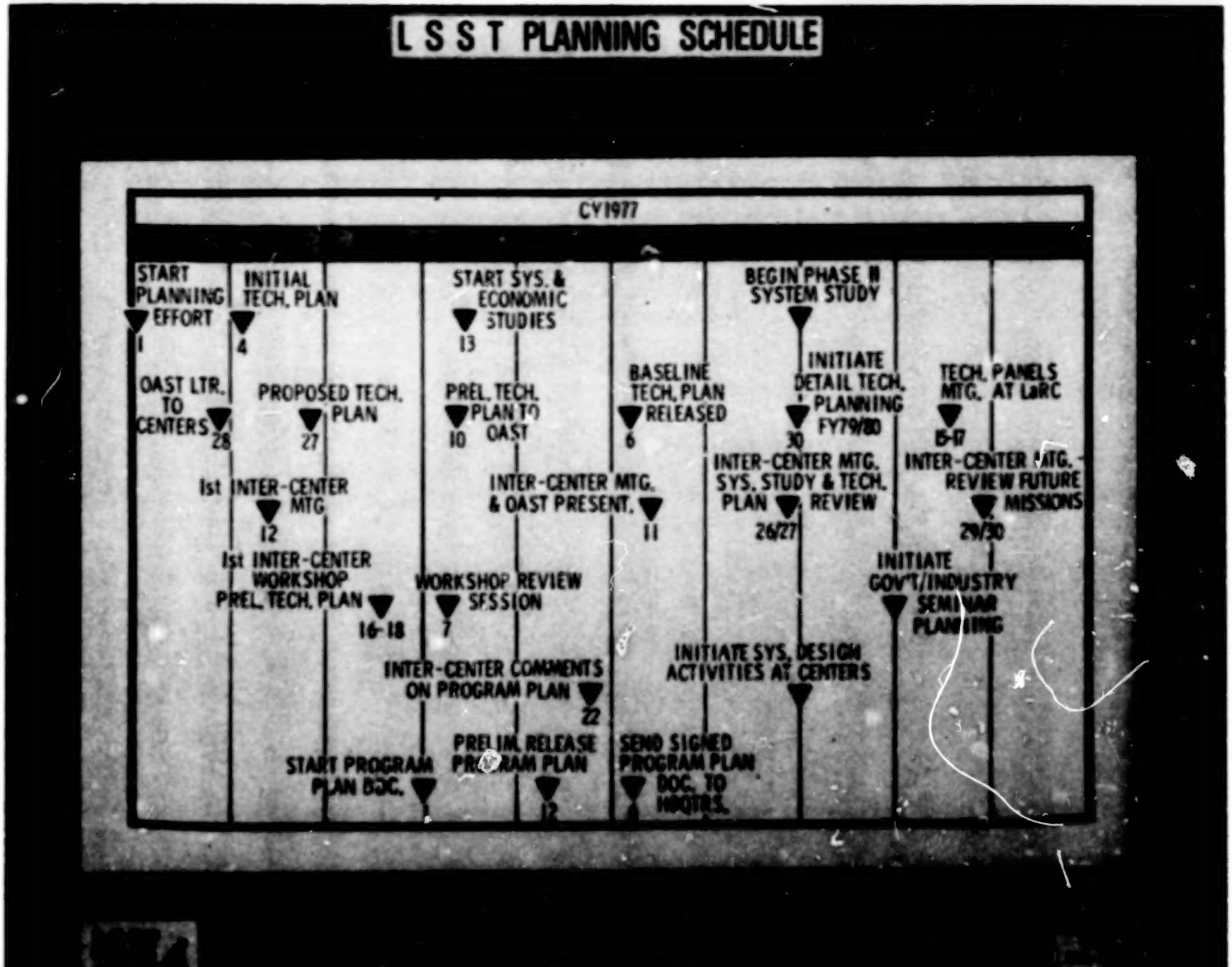
Figure 3

LSST PLANNING SCHEDULE (Figure 4)

The development of advanced technology, to the point of readiness for application, is on the order of 6 to 10 years. With the goal of the LSST Program to provide developed and verified large area space systems technology for missions beginning in 1985 (with earlier "spin-offs" requested), it is important to start the program in FY 79.

Early implementation of the LSST Program to start the development of advanced technology for large area space systems on an integrated basis is both efficient and cost effective. The outputs of this program will not only benefit the civilian space program, but will have applications to DOD programs that require the utilization of large space systems in the same time period.

The LSST Program was initiated as an Inter-Center effort by an OAST letter to the Centers on March 28, 1977. Efforts in CY 1977 were directed towards program planning and involved some systems study effort and a number of meetings of the Inter-Center Working Group to identify and define potential technology development areas.



L S S T PLANNING SCHEDULE
CY1977
START PLANNING EFFORT
1
INITIAL TECH. PLAN
4
START SYS. & ECONOMIC STUDIES
13
BEGIN PHASE II SYSTEM STUDY
OAST LTR. TO CENTERS
28
PROPOSED TECH. PLAN
27
PREL. TECH. PLAN TO OAST
10
BASELINE TECH. PLAN RELEASED
6
INITIATE DETAIL TECH. PLANNING FY79/80
30
TECH. PANELS MTG. AT LaRC
15-17
1st INTER-CENTER MTG
12
INTER-CENTER MTG. & OAST PRESENT.
11
INTER-CENTER MTG. SYS. STUDY & TECH. PLAN REVIEW
26/27
INTER-CENTER MTG. - REVIEW FUTURE MISSIONS
29/30
1st INTER-CENTER WORKSHOP PREL. TECH. PLAN
16-18
WORKSHOP REVIEW SESSION
7
INITIATE GOV'T/INDUSTRY SEMINAR PLANNING
INTER-CENTER COMMENTS ON PROGRAM PLAN
22
INITIATE SYS. DESIGN ACTIVITIES AT CENTERS
START PROGRAM PLAN DOC.
PRELIM. RELEASE PROGRAM PLAN
12
SEND SIGNED PROGRAM PLAN DOC. TO HDQTRS.

Figure 4

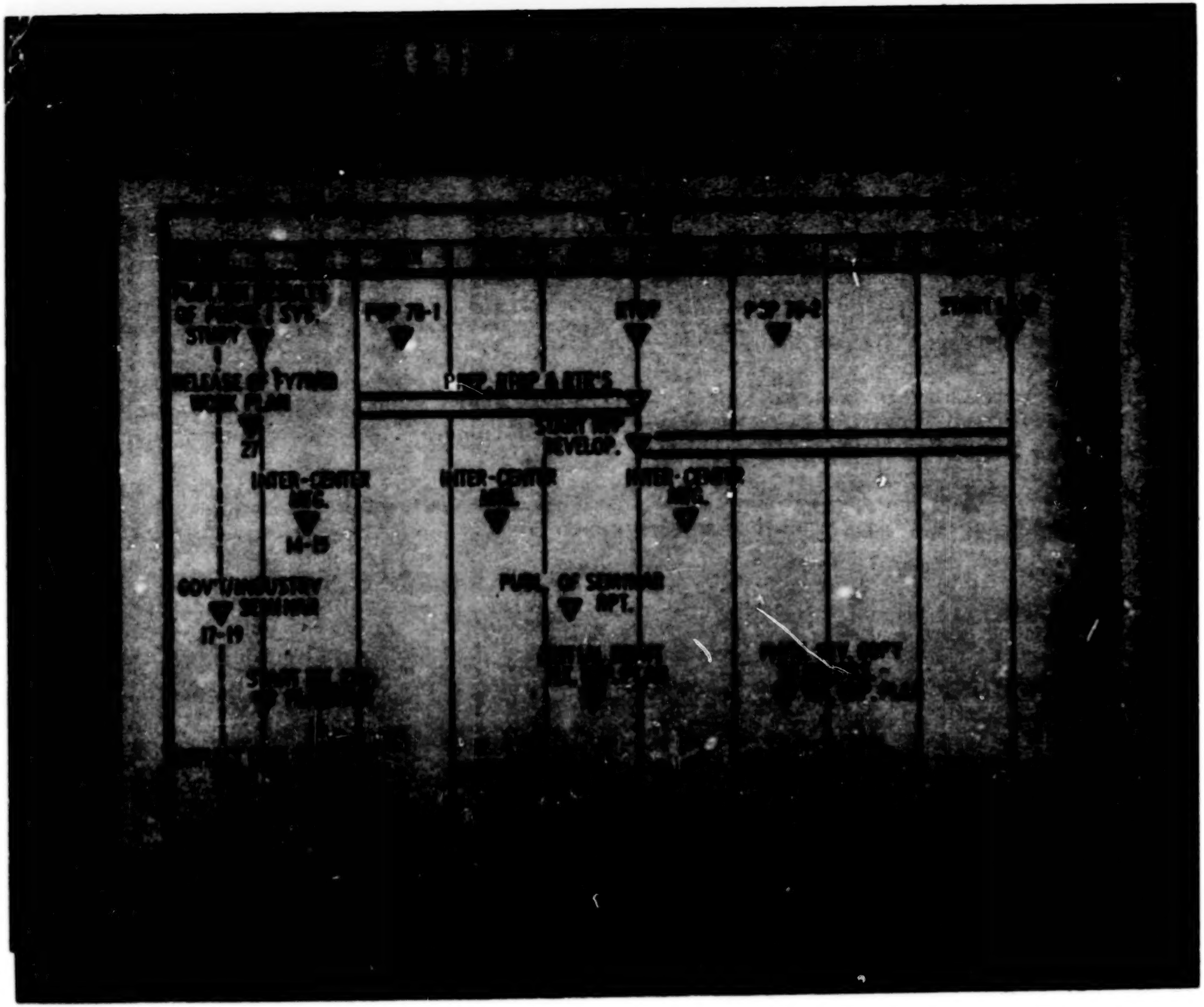
LSST PLANNING SCHEDULE (Cont'd.) (Figure 5)

Efforts in early CY 1978 were directed towards finalizing the FY 1979 Technology Work Plan which contains the division of work and funding among participating centers. The plan was released on January 27, 1978.

A Government/Industry Seminar on Large Space Systems was held at Langley on January 17-19, 1978. This very successful seminar resulted in a free exchange of information and a number of very useful observations and recommendations.

Since the LSST Program will be funded in FY 1979, each participating center must prepare the necessary RTR's starting in March 1978. RTOP's, incorporating inputs from these RTR's will be submitted to NASA Headquarters about June 1, 1978.

Centers which plan to utilize contractors should start preparation of their RFP's by mid-1978 so that contracts can be awarded as early as possible in FY 1979 when LSST funds become available.

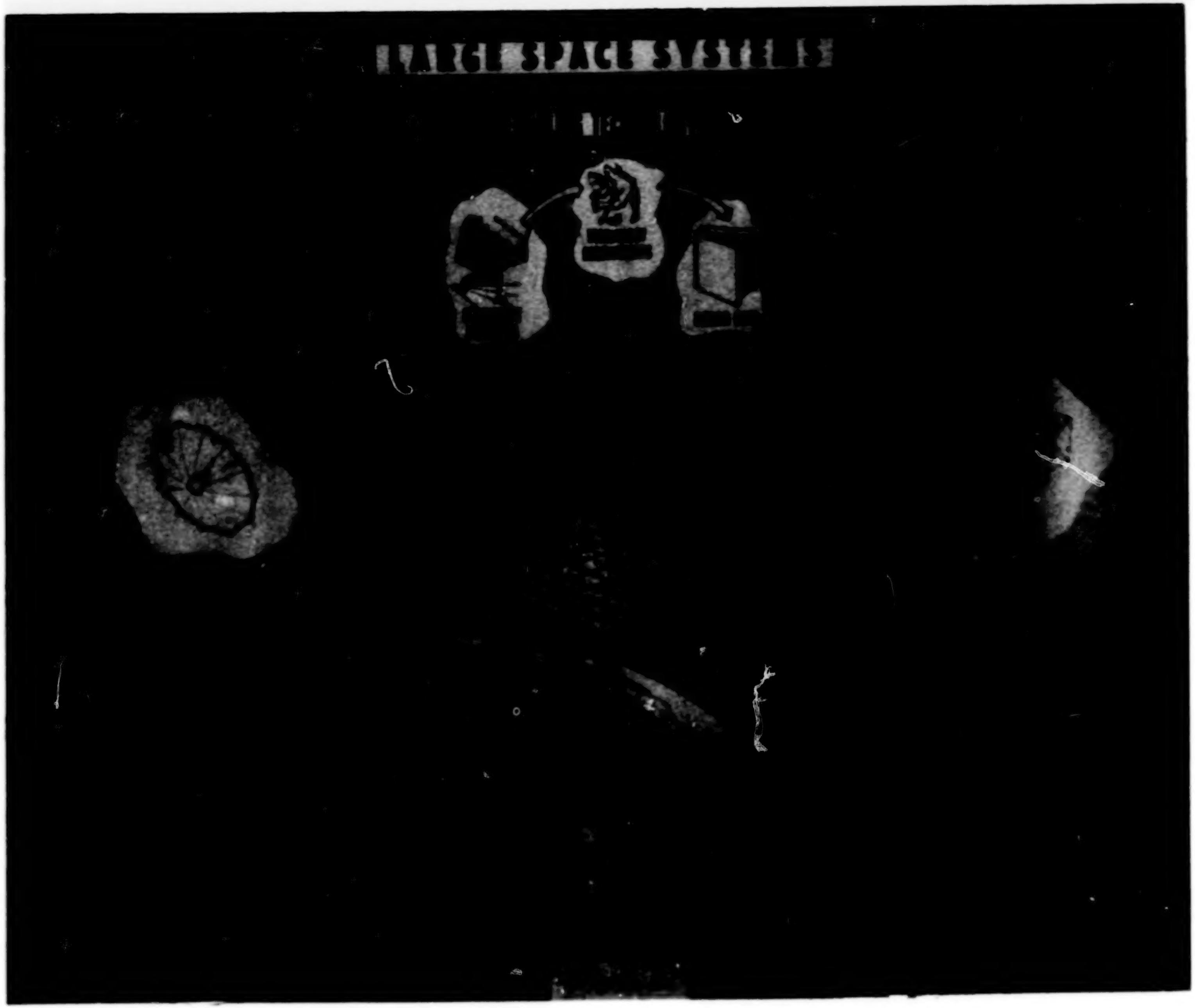

P-SP 70-1
RTOP
P-SP 70-2
DEVELOP.
INTER-CENTER
MTG.
14-15
INTER-CENTER
MTG.
INTER-CENTER
MTG.
GOV'T/INDUSTRY
SEMINAR
17-19
PUBL. OF SEMINAR
RPT.
27

LARGE SPACE SYSTEMS (Figure 6)

Systems Studies

The Systems Area plays a vital role in the LSST Program. The objective of the activities in this area is to assure that all technologies being developed by the LSST Program are properly focused and integrated throughout the life of the program.

Systems studies will be performed to identify the large space systems required to achieve National and NASA goals along with technology development requirements. These identified systems will then be used as focal points, or focus missions, for integrating and continually updating and evaluating the technology being developed in all areas of the program. Economic and scientific benefits will be an important consideration in the selection of LSST focus missions. Large space systems capable of executing focus missions will be selected and definitized to the point where requirements for technology developments are ascertained.


LARGE SPACE SYSTEMS

STRUCTURES TECHNOLOGY (Figure 7)

Structural Concepts-Erectable

The Erectable Structural Concepts area will provide the technology needed to design, fabricate, and assemble large, structurally efficient, low-cost structural systems ranging in size from 50 to 1000 meters. A major target of this technology area is to define, develop, and evaluate relatively low mass structural concepts that can be fabricated on Earth, efficiently packaged into the Shuttle, transported to orbit, and assembled in a timely fashion either from the Shuttle or from an auxiliary construction platform.

Structural Concepts-Deployable Platform

The thrust of this technology area is to define, develop, and evaluate low mass structural concepts that can be fabricated, assembled, and packaged on Earth, stowed in the Shuttle bay, and automatically deployed in space into a structure 100 meters or larger in size. In order to examine the possibility of constructing structures much larger than 100 meters using deployable modules, methods of coupling deployed modules will be investigated.

Space Fabrication

The applicability and cost effectiveness of using space fabrication techniques for building large platforms in the foci missions will be the subject of system studies in the initial phase of the program. Space fabrication techniques involve fabrication and assembly of structural members from densely packaged, pre-processed material. Structural concepts, materials, and equipment technology will be added to the LSST Program based on results of the system studies.

Structural Concepts-Deployable Reflectors

The principal target of this technology area is to define, d[illegible]lop and evaluate low mass reflector concepts that can be fabricated, assembled, and packaged on Earth, st[illegible]d in the Shuttle, and automatically deployed into shaped surfaces 30 to 300 meters in size with rms surface accuracies consistent with the foci of the LSST Program. Methods must be developed to measure the shape of the surface and appropriate mechanisms designed to change the surface to the required shape.

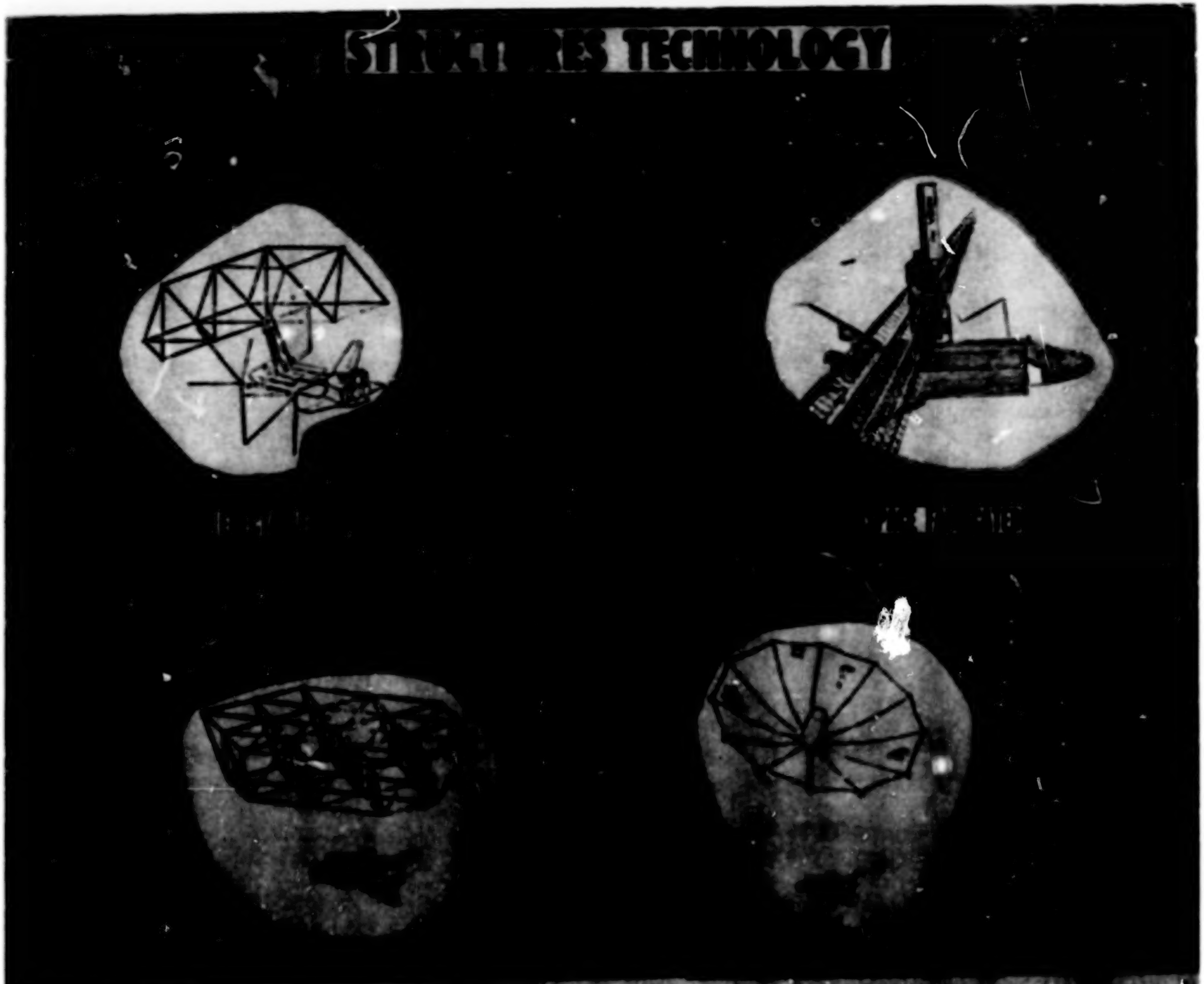


Figure 7

COMPLEMENTARY TECHNOLOGY AREAS (Figure 8)

Analysis and Integrated Design

Methods for the efficient prediction of electromagnetic properties of large reflectors will be obtained by the development of cost effective electromagnetic field prediction analyses for scanning type antennas and multi-beam continous surface reflector type antennas. The tools needed for the design of appropriate control systems will be obtained through development of analysis techniques for maximum performance through placement of sensors and actuators and synthesis techniques of control laws and structural characterization. Advanced analytical tools will also be developed for the prediction of structural loads and distortions.

Control Systems

Work in the Control Systems area will provide the technology required to design, fabricate, and evaluate three kinds of control systems: shape, attitude, and orbit stationkeeping.

Electronics

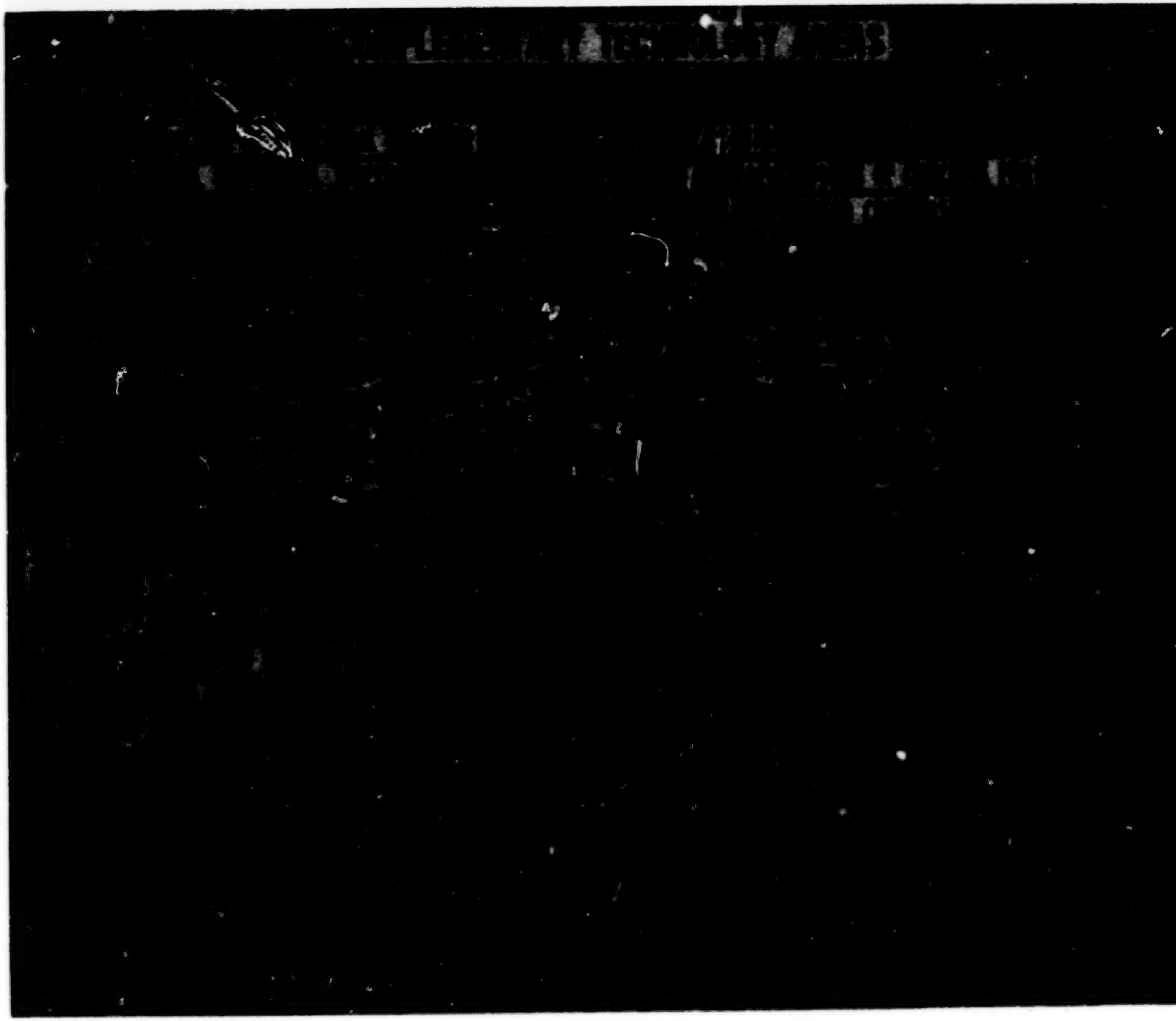
Electrical subsystems will be used for active surface control as well as in providing housekeeping data that must be used in verifying system performance. Therefore, techniques will be developed for signal conditioning, data acquisition, and data transfer.

Power must be distributed throughout a large space system. Both centralized and distributed power systems will be evaluated and trade-off studies conducted. In addition, studies and tests will be performed to determine the feasibility of using the structure itself as a conductor for distributing power and thereby reducing the mass of the overall system.

Techniques will be investigated that will reduce data channel interference and multipaction effects. Such interference could severely degrade the data handling capacity of the instrumentation subsystem on board the large space system.

Advanced Materials and Joining

Accelerated laboratory testing procedures and associated analyses will be developed to predict lifetime of both metals and composites in the 30 year range. Long life, dimensionally stable polymeric matrix composites will be developed. If necessary, metal and glass matrix composites will be developed. Thin gauge, low mass structural alloys will also be investigated. To solve some of the serious thermal problems, advanced thermal control concepts such as integral control surface and heat pipes technology will be examined.



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TECHNOLOGY NEEDS AND OPPORTUNITIES FOR FUTURE NASA MISSIONS

S. R. SADIN

Presented at:

Government/Industry Seminar on
Large Space Systems Technology

Langley Research Center
Hampton, Va. 23665

17 JANUARY 1978

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This paper deals with the process of forecasting, both of NASA's future needs and missions and of the trends of the technologies relevant to the projected needs. By comparing the two forecasts, one may identify potential gaps or voids in the availability of technology. Once having identified such gaps and voids, one may prioritize and weigh the technological needs to generate plans for investment in research and development to help enable the potential missions to take place. In particular, this paper deals with those technology issues (e.g., structures, control of large space structures, materials for large space systems, automation) and potential NASA missions that relate to large space systems technology.

(Figure 1)

This scene depicts the impact of technological pollution at the turn of the last century, a technological pollution introduced by the first big step in communications--the telephone. As indicated on the chart, underground cables and microwave relays came along to provide a technological solution to eliminate the visual pollution of urban streets by pole lines. Technology has continued to provide better and better answers to this kind of pollution, including communication satellites and the use of fiber optics as a major advance over the kinds of cables that first replaced the wires shown in this picture.

TELEPHONE WIRES
IN NEW YORK IN 1890

Pole lines in New York at a time when there were about 8,000 telephones on Manhattan Island. Without the research that produced underground cables, and then later microwave relays, the growth of the telephone system would have stopped.

Source: The Bell Telephone System, A.W. Page, 1940

Figure 1

(Figure 2)

On this chart we can see the beginnings of a new and apparent form of technological pollution being introduced by communication satellites in geo-synchronous orbit. These are only the communication satellites for the year 1980. If one were to project ten or twenty years beyond 1980, geosynchronous orbit would be, by comparison, as severely cluttered and polluted as our cities were with telephone wires a hundred years earlier. Fortunately, technology is once again evolving to provide solutions that will alleviate this apparent pollution. These solutions derive from the capabilities of basic electronics and the capabilities of future large space systems.

COMMUNICATION SATELLITES IN GEOSTATIONARY ORBITS BY 1980

SOURCE: Economic and Policy Problems in Satellite Communications, Pelton and Snow, 1977

*Not operational or in service as of September 1977

Figure 2

(Figure 3)

It is apparent from this figure that the phenomenal growth that we have witnessed in electronics technology in recent years is evidenced not only in performance (as demonstrated here under the measure of speed of operations) but also in terms of reduction of cost and increased reliability. All of these capabilities are progressing at the remarkable exponential rate of doubling every one to three years. This growth is reflected in almost any measure of computer performance, cost, or reliability. Translated into the communications satellites of the future, this means that we can look forward to a highly complex payload capability for communications satellites. The communication gear itself will be capable of being built at very low cost and operating without the need for repair for very extensive periods, perhaps five to ten years. In fact, five to ten year missions are the predicted technological obsolescence period for electronics. We have a fortuitous capability that is blossoming out of the basic electronic business.

Electronics Technology Progress

PAST AND FUTURE

- Average rate of growth = doubling every 1½ to 3 years

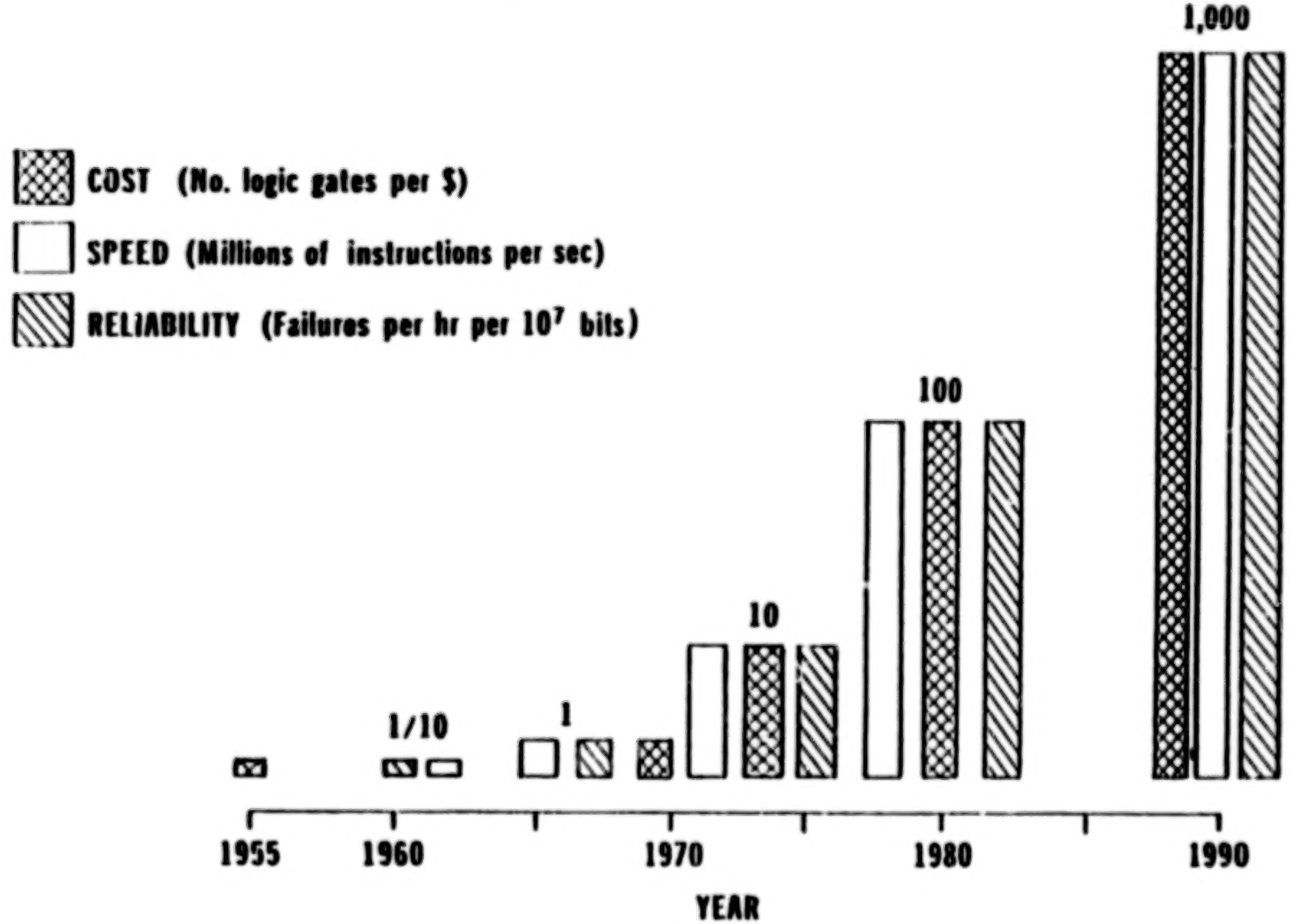


Figure 3

28

(Figure 4)

Basic electronics progress serves as the driver to future programs as shown in this example of a large multipurpose communication platform in geostationary orbit that can serve many user needs. This vehicle capability is a challenge to the community represented by the persons in attendance at this meeting here at Langley. It provides for the care and feeding of multiple payloads in earth orbit, payloads of a complex, sophisticated, and reliable nature able to communicate with simple user-oriented ground terminals on earth. This foretells an era of so-called complexity inversion in which the orginal ground rules around which the space program evolved are radically altered. Originally, the philosophy governing space systems was to place the smallest number of active parts in space, and leave the complexity on the ground. Because of the growing interest and need for user participation, and the growth of technology, we are beginning to see this inversion evolve.

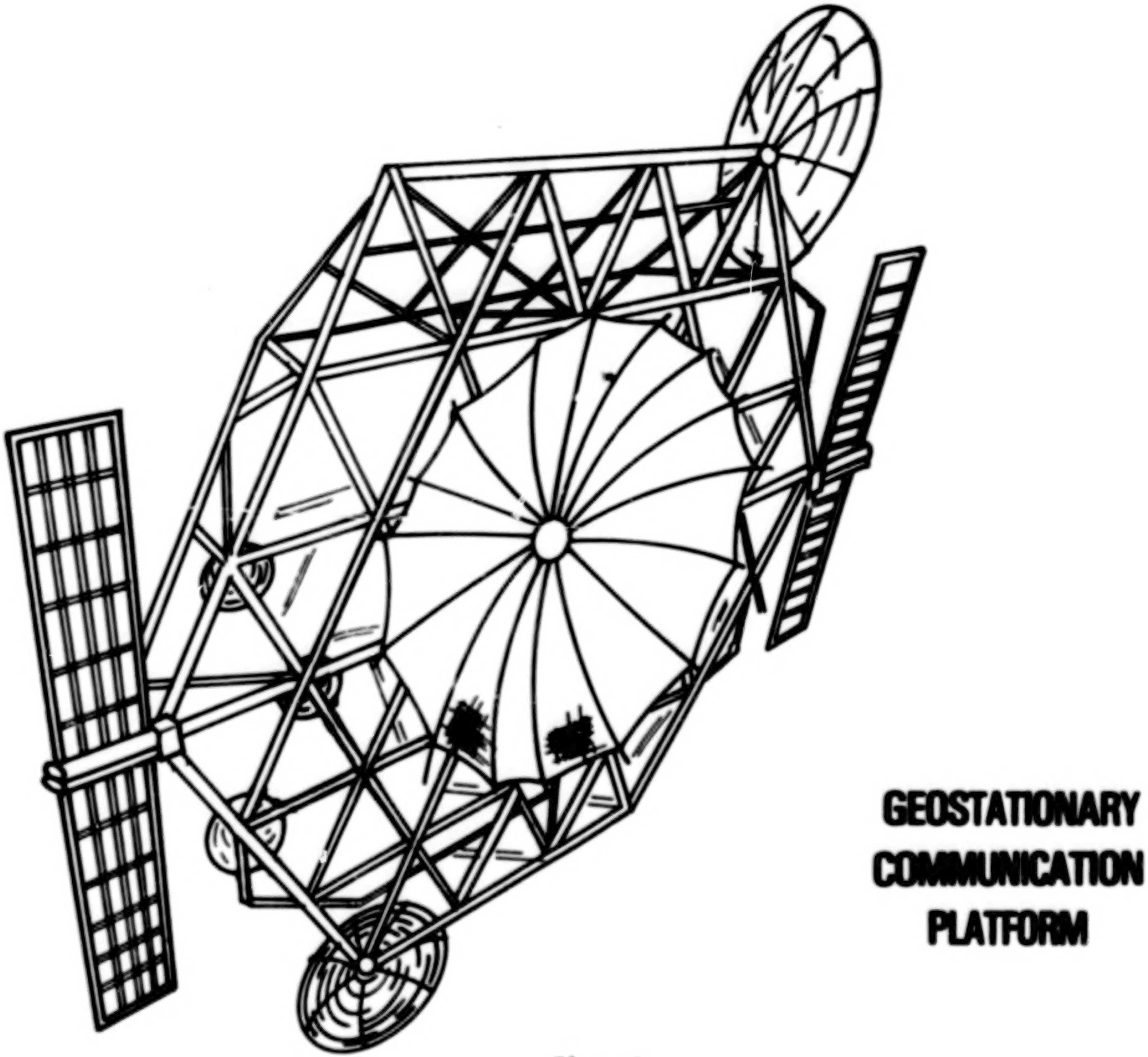


Figure 4

(Figure 5)

30 Let us begin now to look at some of the specific challenges in technology growth that are going to be faced by large space systems of the future. This figure shows both the historic trend and a forecast of the physical size of spacecraft systems. We can see that while today's space system size capability is only in the low tens-of-meters, in the 1980s we will be moving into the area of hundreds-of-meters, and probably in the 1990s into a thousands-of-meters' capability as evidenced by the programs shown in the figure. The 25-kW power system is a program that is considered part of NASA's current five-year planning interests, whereas the programs identified as electronic mail systems and satellite power stations are representative models of long-range future potential systems and are not part of the Agency's committed five-year plan.

SPACE SYSTEM TECHNOLOGY FORECAST – SPACECRAFT SYSTEMS

- **LARGE SPACE STRUCTURES**

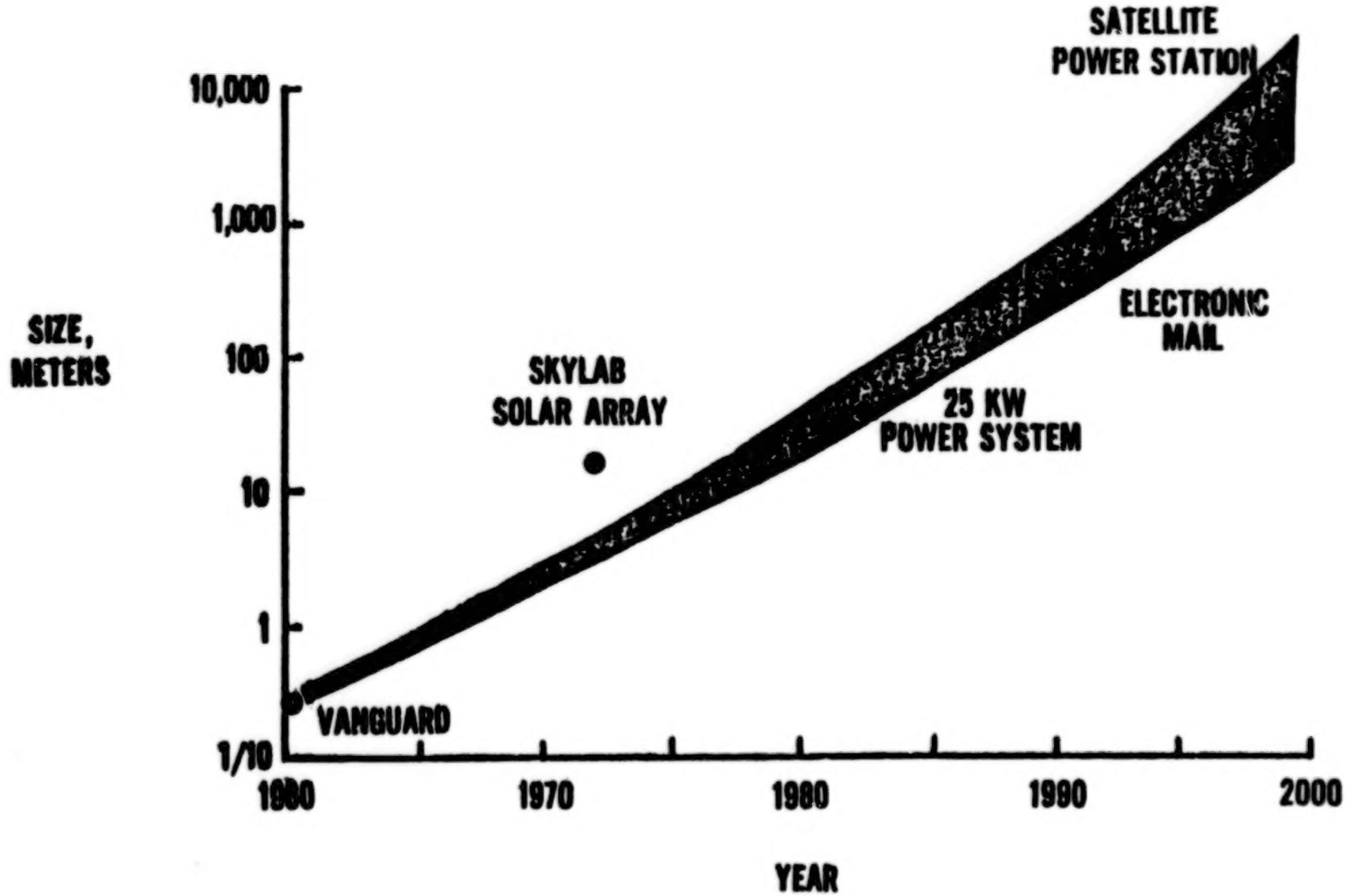


Figure 5

31

32

(Figure 6)

In order to forecast or predict future Agency needs, we have generated a technology model which is described on the VuGraph. The model is an aid to technology planning and certainly not a policy document, an Agency hard plan, or even an attempt to accurately predict the future. The near-term mission opportunities deal with real programs that are part of our Agency's planning, while the far-term mission opportunities are hypothetical future programs. The model purpose is stated in several ways, but basically the primary purpose of the document is to serve as a part of the OAST planning process.

OAST Space Systems Technology Model

- MODEL <u>IS</u> A SET OF GENERIC SYSTEMS
  - SELECTED TO DRIVE CRITICAL TECHNOLOGIES
  - DESIGNED TO PROVIDE REALISTIC CHALLENGES FOR TECHNOLOGY AREAS
- MODEL <u>IS NOT</u>
  - PREDICTING THE FUTURE
  - STATING NASA/OAST POLICY OR PLANS
- DIVIDED INTO NEAR & FAR TERM MISSION OPPORTUNITIES
- PURPOSES
  - ASSIST IN DEVELOPING TECHNOLOGY PROGRAM OPTIONS
  - IDENTIFY MAJOR TECHNOLOGY AREAS REQUIRING CONCENTRATED EFFORTS
  - SERVE AS AN EVALUATION CRITERIA FOR CURRENT TECHNOLOGY PROGRAMS
- PART OF THE OAST TECHNOLOGY PROGRAM PLANNING PROCESS

Figure 6

(Figure 7)

This chart indicates that, beginning with the major Agency exercise of the Outlook for Space one year long effort, there has been a series of far-reaching program studies and workshops to help us generate the elements of this technology model.

OAST Space Systems Technology Model

- KEY DATA SOURCES
  - NASA OUTLOOK FOR SPACE
  - OAST SPACE THEME WORKSHOP
  - NASA FIVE YEAR PLAN
  - NASA ADVANCED STUDIES REPORTS
  - NASA PROGRAM OFFICES
  - OAST SPACE DIRECTORS
  - NASA CENTER TECHNOLOGISTS
  - GRC TECHNICAL STAFF

Figure 7

(Figure 8)

I would like to show you representative near-term Exploration missions as they were published in the 1977 Agency Five-Year Plan document. This document is in the process of revision and will deal with the FY 1979 realities as being processed in the budgetary Congressional approval cycles at this time. If anything, the missions shown here , which were conceived in the FY 1978 planning cycle, are optimistic. Specifically, one would not expect to see as many missions as are shown here. Note that, while a rendezvous with a comet is under study, our plans for a Halley's comet rendezvous no longer exist so the date of any comet mission is further off. The pinhole satellite has been flagged as being impacted by the requirements of large space systems.

OAST Space Systems Technology Model

NEAR TERM MISSIONS

EXPLORATION OF THE UNIVERSE

1. Space Telescope
2. Out-of-Ecliptic Mission
3. Solar Maximum Mission Reflight
4. Spacelab Multiuser Instrument Program
5. Solar Probe
6. Origin of Plasmas in the Earth's Neighborhood (OPEN)
7. Synoptic Troposphere and Terraspace Environment Satellite
8. * Pinhole Satellite
9. Solar Mesosphere Explorer
10. Active Magnetospheric Particle Tracer Experiment
11. Cosmic/X-Ray Observatory
12. HEAO-B
13. HEAO-C
14. Advanced X-Ray Astrophysical Facility
15. Cosmic Background Explorer
16. Gamma Ray Observatory
17. Large Area Moderate Angular Resolution X-Ray Array
18. Lunar Polar Orbiter
19. Jupiter Orbiter/Probe
20. Viking Mobile Lander
21. Comet Rendezvous
22. Venus Orbital Imaging Radar
23. Mars Surface Sample Return
24. Saturn Orbiter / Probe
25. Extreme Ultraviolet Explorer

* Large Space Systems Technology Requirements

Figure 8

37

CONTENTS

(Figure 9)

As an example of treating the technology in near-term missions, we have the pinhole satellite system depicted in this sketch. The reason we have selected this as a large space system example is because of the separate mask which is a fairly large structural element (~50 m) in space. Basically, this satellite consists of two parts: (1) a large, opaque mask (made of lead or other dense material) with a random pinhole array, and (2) a detector array located up to one kilometer away from the mask. Together they form a space telescope which will be pointed at the sun to detect hard x-rays ranging from 10-100 keV.

PINHOLE

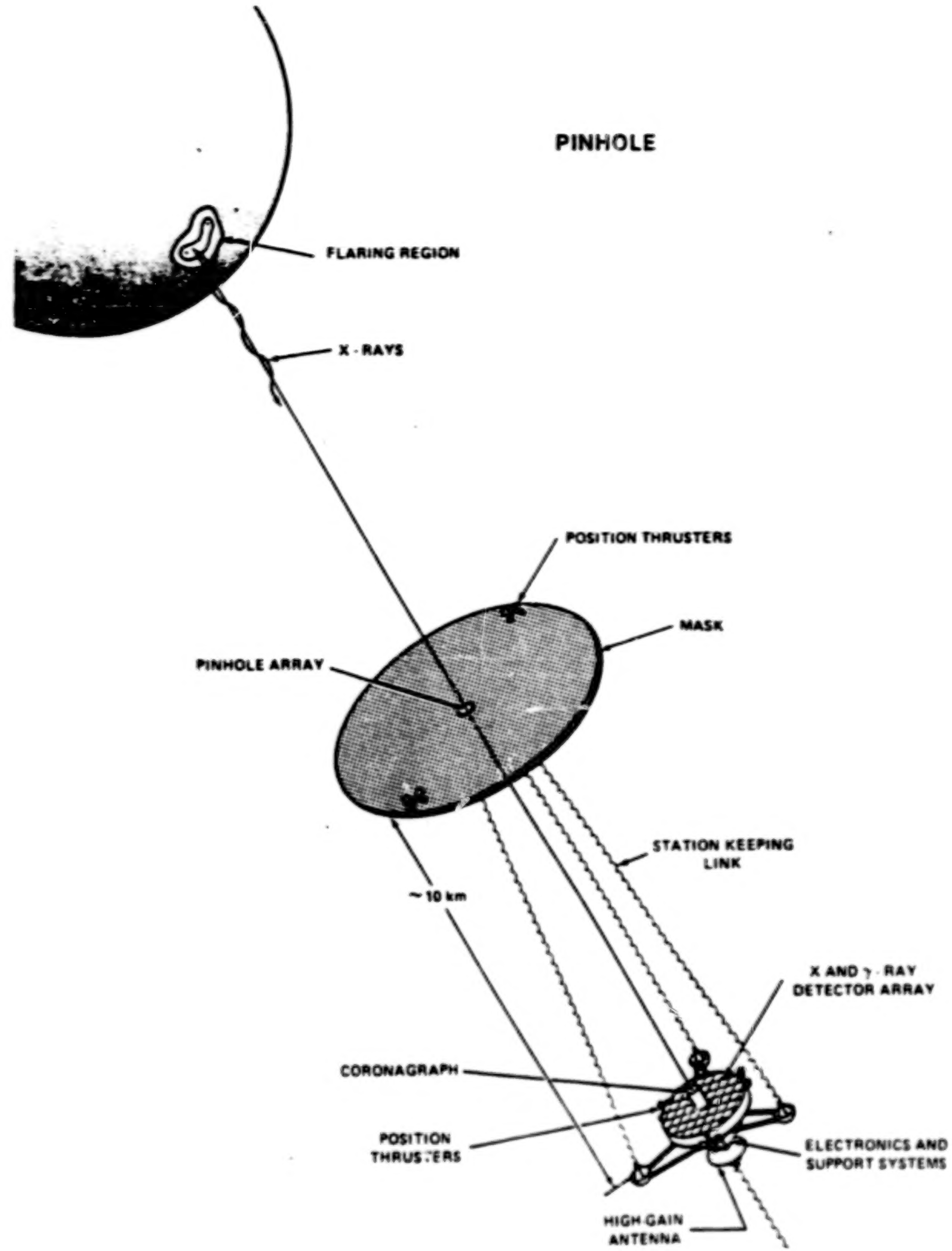
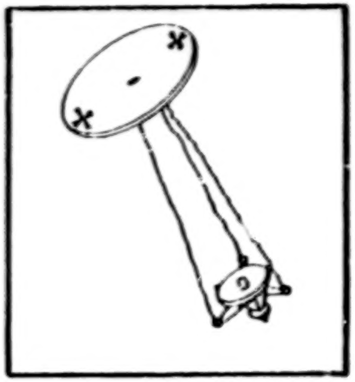


Figure 9

(Figure 10)

This chart shows the first page of the pinhole satellite description as it appears in the OAST Space Systems Technology model. Each of the near-term Exploration missions listed on the earlier VuGraph is similarly treated in the model with a brief description, conceptual sketch, and a list of technology needs, capability measures, and desired values.



MISSION MODEL - SAMPLE

PINHOLE SATELLITE

DESCRIPTION: The Pinhole Satellite is a space telescope consisting of a large, opaque mask (made of lead or other dense material) with a random pinhole array and a detector array located up to 1-km away from the mask. This telescope will be pointed at the sun to detect hard X-rays ranging from 10-100 keV. There are several concepts for this satellite, including: both mask and detector array free flying; a shuttle-based mask and a free flying detector array; a free flying mask and a shuttle-based detector array.

PRIMARY TECHNOLOGY NEEDS	CAPABILITY MEASURE	DESIRED VALUE
Large space structures assembly	Size	≥50 m
X-Ray Detector Assembly		
Imaging proportional counter	Size	2 m × 2 m
	Weight	250 kg
	Power	320 W
	Resolution	1 to 5 mm @ 1 km
Alignment prism	Weight	2 kg
Beacon (alternative to prism)	Weight	25 kg
	Power	1000 W
Alignment Base		
Sun sensor	Detector size	10 × 10 × 3 cm
	Electronics size	20 × 12 × 7 cm
	Weight	4 kg
	Power	2 W
	Stability	0.5 arc sec
	Accuracy	1.0 arc min

Figure 10

(Figure 11)

The first chart of near-term mission opportunities dealt only with the Exploration program. This chart lists the balance of systems, including Global Services, Utilization of Space, Environment and Space Transportation mission opportunities. The starred systems are those that are impacted by the requirements of large space systems and are the ones from which my examples will be selected.

OAST Space Systems Technology Model

NEAR TERM MISSIONS (Cont.)

GLOBAL SERVICES

26. Multimission Modular Spacecraft
27. SEASAT-B
28. Environmental Monitoring Satellite
*29. Global Communications Land Mobile Services
30. Stormsat-A
31. Geodetic Survey Satellite
*32. Public Service Communications Satellite
33. TIROS-O
*34. Geostationary Platform
*35. Molecular Wake Shield

UTILIZATION OF THE SPACE ENVIRONMENT

36. Teleoperator Orbiter Bay Experiment
37. Advanced Spacelab Space Processing Payloads
38. Space Manufacturing Module
*39. 250-kW Power Module
40. 25-kW Power Module

SPACE TRANSPORTATION SYSTEMS

41. Spin-Stabilized Upper Stage
*42. Solar Electric Propulsion Stage
43. Orbital Transfer Vehicle

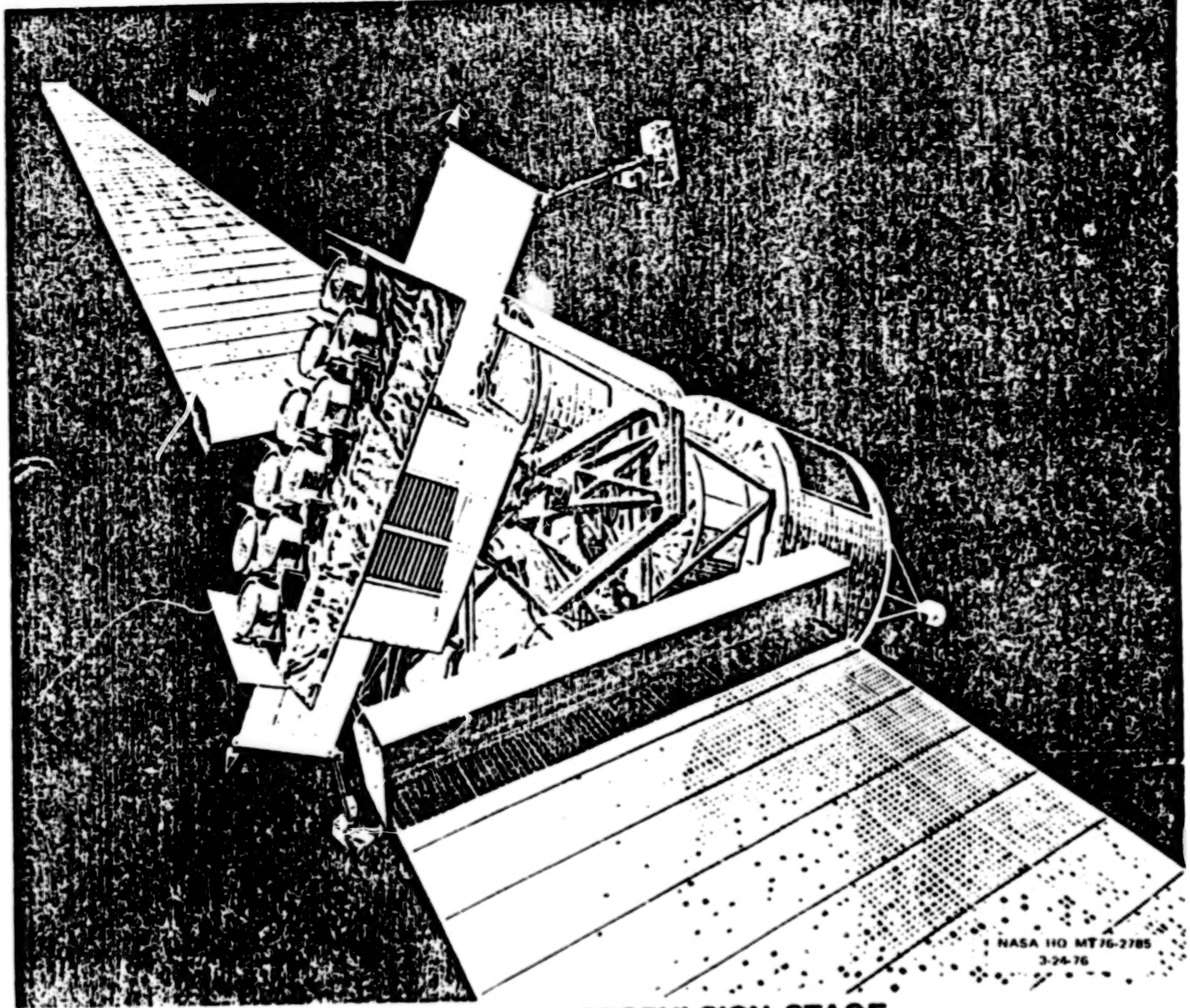
* Large Space Systems Technology Requirements

Figure 11

(Figure 12)

44

To illustrate a mission opportunity in the area of transportation systems, here is an artist's rendition of a solar electric propulsion stage (SEPS). Initially, SEPS would evolve from the requirements of Exploration programs such as the comet rendezvous mission. Later uses of the SEPS could be in support of Global Services or Application missions to provide near earth, orbit-to-orbit positioning capability.



SOLAR ELECTRIC PROPULSION STAGE

Figure 12

45

(Figure 13)

Turning now to the far-term model, we have displayed on this chart twenty-three different far-term conceptual missions. It is important again to appreciate that these are not proposals for real missions nor are they necessarily intended to predict what will actually happen. They are only models that allow us to bound the future in terms of technology requirements and as such they are a representative segment of potential future programs. Once again, mission opportunities impacted by large space systems technology requirements are indicated.

OAST Space Systems Technology Model

FAR TERM MISSIONS

EXPLORATION OF THE UNIVERSE

* 1 AUTOMATED PLANETARY STATION
  2 LARGE EARTH ORBITAL SOLAR OBSERVATORY
  3 ASTROPHYSICS SPACE LABORATORY
  4 ATMOSPHERIC PHYSICS LABORATORY
* 5 SPACE-BASED RADIO TELESCOPE

UTILIZATION OF THE SPACE ENVIRONMENT

* 15 SPACE STATION
  16 NUCLEAR WASTE DISPOSAL SYSTEM
* 17 TELEOPERATOR VEHICLE SYSTEM
  18 LUNAR BASE

GLOBAL SERVICES

  6 LARGE-SCALE ALL-WEATHER SURVEY SYSTEM
  7 HIGH-RESOLUTION SEA SURVEY SYSTEM
* 8 GEOLOGICAL MAPPING SYSTEM
  9 EARTH ENERGY MONITORING SYSTEM
* 10 GLOBAL COMMUNICATIONS SYSTEM
* 11 GLOBAL NAVIGATION SYSTEM
  12 GLOBAL CROP INVENTORY SYSTEM
  13 DISASTER WARNING SYSTEM
* 14 SPACE POWER SYSTEM

SPACE TRANSPORTATION SYSTEMS

* 19 PRIORITY LAUNCH VEHICLE
* 20 HEAVY-LIFT LAUNCH VEHICLE
  21 PRIORITY ORBITAL TRANSFER VEHICLE
  22 CARGO ORBITAL TRANSFER VEHICLE
* 23 ORBITAL ESCAPE VEHICLE

* Large Space Systems Technology Requirements

Figure 13

(Figure 14)

As an example of an Exploration mission opportunity, this figure depicts a very large space-based radio telescope that could conduct radio astronomy experiments and collect electromagnetic radiation from space to detect coherent signals derived from sources of intelligent life in the universe. The system depicted is a kilometer-sized [illegible]ture. At the present, we are studying this application for approximate sizes 300 m. One of the papers in this session ("Technology for Accurate Surface and Attitude Control of a Large Spaceborne Antenna and Microwave System") will deal in some greater depth with a funded study of such a 300-m radio astronomy telescope.

LARGE SPACE RADIO TELESCOPE C[illegible]

Figure 14

50

(Figure 15)

This chart deals with one of the major problems in going to a very large flexible antenna structure in space--the need to maintain figure control. Surface precision is shown here in the nondimensional units of diameter divided by the RMS surface error. This graph has an earlier forecast shown in the shaded area. This earlier prediction showed that we had to increase surface precision about one order of magnitude in order to meet future program needs. The study of the 300-m telescope yielded the requirements shown by the vertical black bar which indicates that the earlier prediction underestimated potential needs. This example, which is admittedly a severe test case, requires as much as three orders of magnitude improvement in figure control. All of this, of course, implies the use of active control. The study investigators describe this 300-m control problem as equivalent to trying to maintain figure control on a set of wet noodles.

Space System Technology Forecast

- SURFACE PRECISION

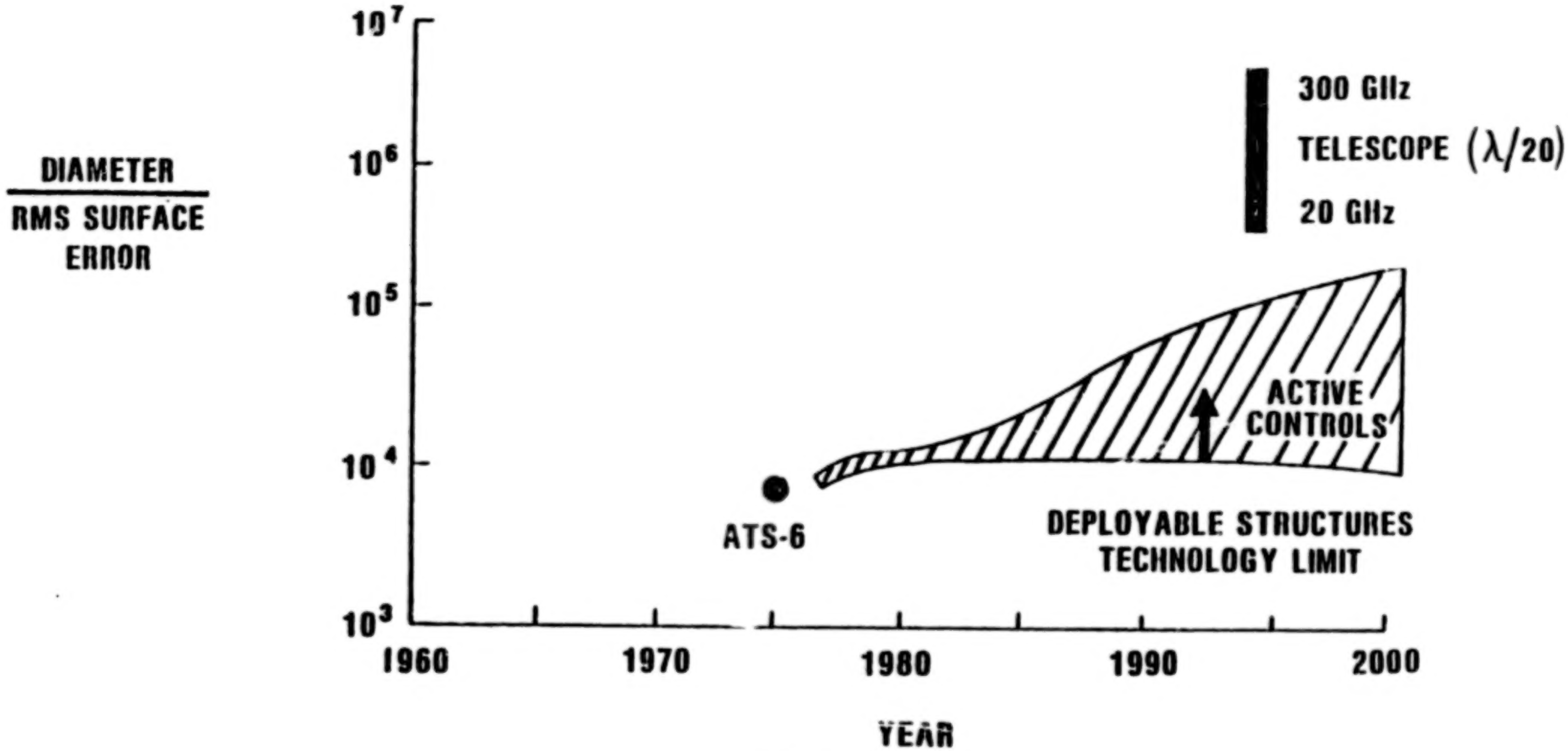


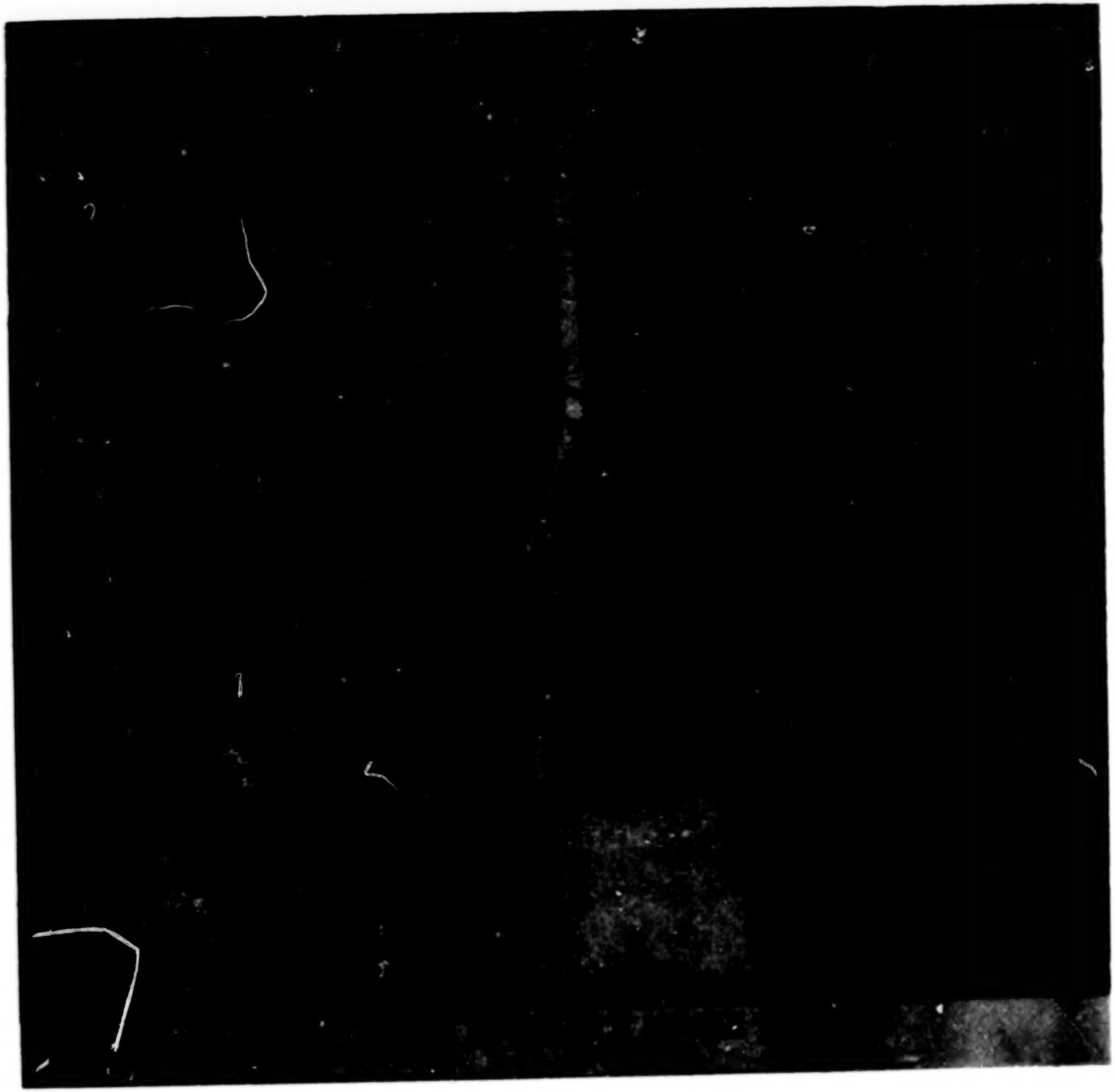
Figure 15

51

52

(Figure 16)

This is the heavy-lift launch vehicle. The concept illustrated is the ballistic-ballistic option where both stages are fully reusable. If we move toward the delivery of massive payloads for orbital assembly of large space systems, we will require very large cargo transports of this type. Transportation systems of this scale impose, in turn, their own requirements on the technological capability in large space systems.



53

(Figure 17)

As shown in this forecast, a shuttle-derived fully reusable cargo system should reduce the transportation costs to low earth orbit from the current hundreds of dollars per kilogram for the space shuttle to the tens of dollars per kilogram range by the end of the century. Developments in technology that will be crucial to achieving this reduction include the use of composites to reduce weight, improvements in propulsion, and significant breakthroughs in operation costs through automated processes.

SPACE SYSTEM TECHNOLOGY FORECASTS – TRANSPORTATION SYSTEMS

- EARTH-TO-LOW-EARTH ORBIT TRANSPORTATION COSTS

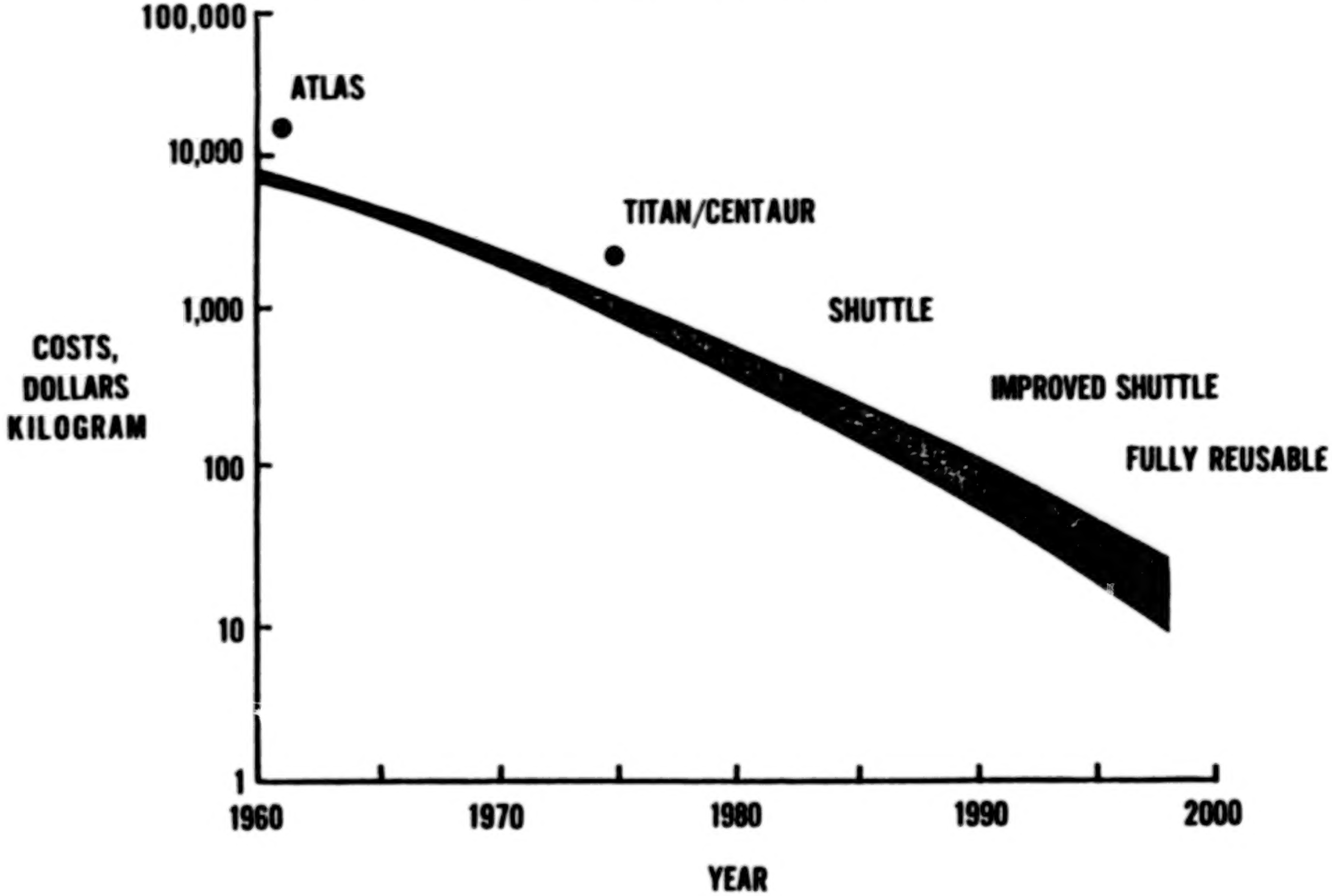


Figure 17

(Figure 18)

In addition to concerns about low unit cargo costs measured on a per kilogram basis, we will require transportation systems that optimize on the basis of unit cost per flight instead of unit weight. A single-stage-to-orbit, shown here as a second generation reusable space booster, is designed for smaller payloads where the concern is rapid access to space with the minimum cost for any single flight. This is in contrast to the heavy-lift system where one would take time to accumulate large quantities of cargo in order to keep the unit cost per pound to orbit down. Similar discipline needs discussed previously for heavy-lift systems are critical to SSTO development.

2nd GENERATION REUSABLE SPACE BOOSTER

Figure 18

(Figure 19)

This figure depicts the technology trend in weight reduction. The feasibility of a single-stage-to-orbit vehicle depends heavily upon this forecast reduction of unit structural weight. Advances in composites, structures, and metallic fully reusable thermal protection and in control-configured design techniques may be expected to contribute significantly to this forecast.

Space System Technology Forecast

● STRUCTURAL WEIGHT REDUCTION FOR SPACE SYSTEMS

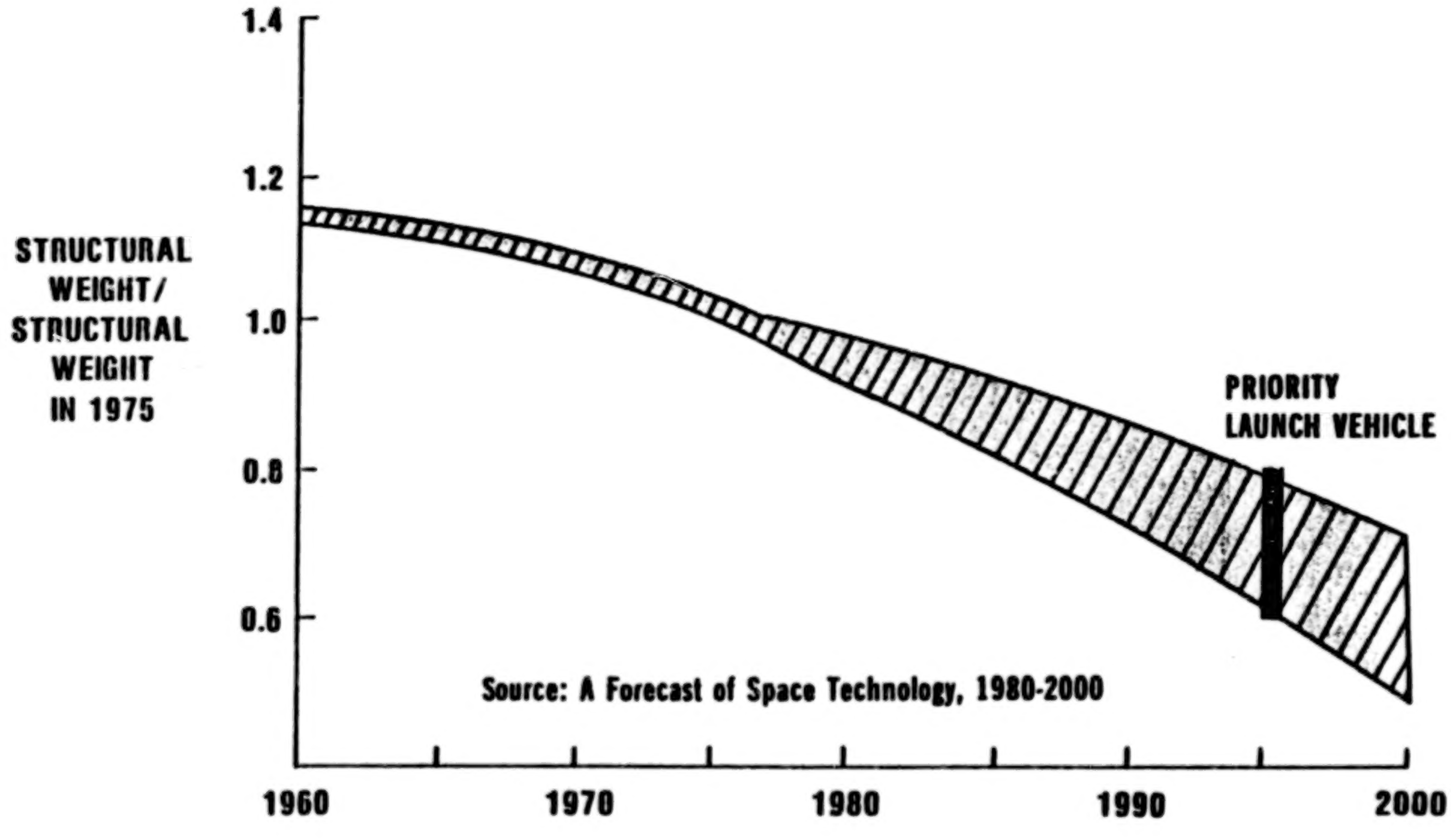


Figure 19

(Figure 20)

This figure displays a concept for Global Services deriving from a Langley Research Center study. The concept envisions large space platforms providing multi-purpose or multimission space services capability. The platform at the lower right is a collection of multimission antennas useful for collecting information from terrestrial sources, or for relaying information from one terrestrial source to another. In the center is a central power plant that could beam power to the antenna platforms. As spacecraft systems grow, the requirements for power expand very rapidly. The possibility also exists for centralizing information processing. These very large platforms serve as a useful guide for planning technology for future large space systems.

FUTURE SPACE SYSTEM OPPORTUNITIES

- MULTIPURPOSE SPACE SERVICES PLATFORM

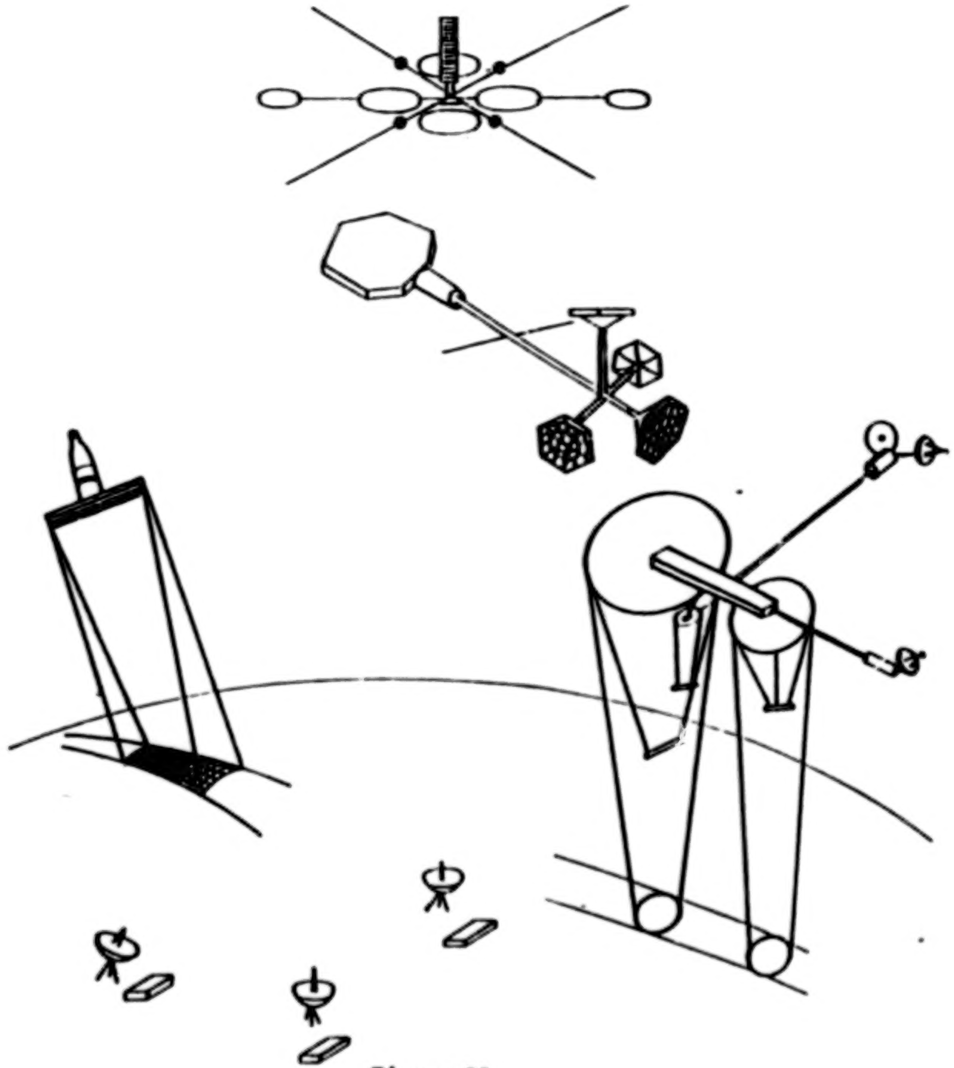


Figure 20

61

(Figure 21)

62

This chart illustrates the energy usage of space systems (in kilowatt-hours) and clearly shows that, including Skylab, the accumulated energy consumption of all our past space programs does not begin to satisfy the needs of many large systems forecast for the future. Future space systems are going to have to develop power capabilities orders of magnitude greater than has been done to date. The implications of this are far-reaching. It is not possible to continue to produce space power systems at the same unit cost per kilowatt-hour that we have in the past. The space program could not support the economic demands of such power systems. Major breakthroughs will have to come in terms of the cost of power systems or in terms of energy sources, a problem very similar to the one that we face here on earth. Unquestionably, the impact of large space systems and in particular large structures and materials on this power problem is significant. Power systems themselves may constitute a major driver to structures and materials technology.

Note that nowhere on this curve are the power requirements for a satellite power station indicated. Those requirements fall above the top of the page.

SPACE SYSTEM TECHNOLOGY FORECAST – POWER SYSTEMS

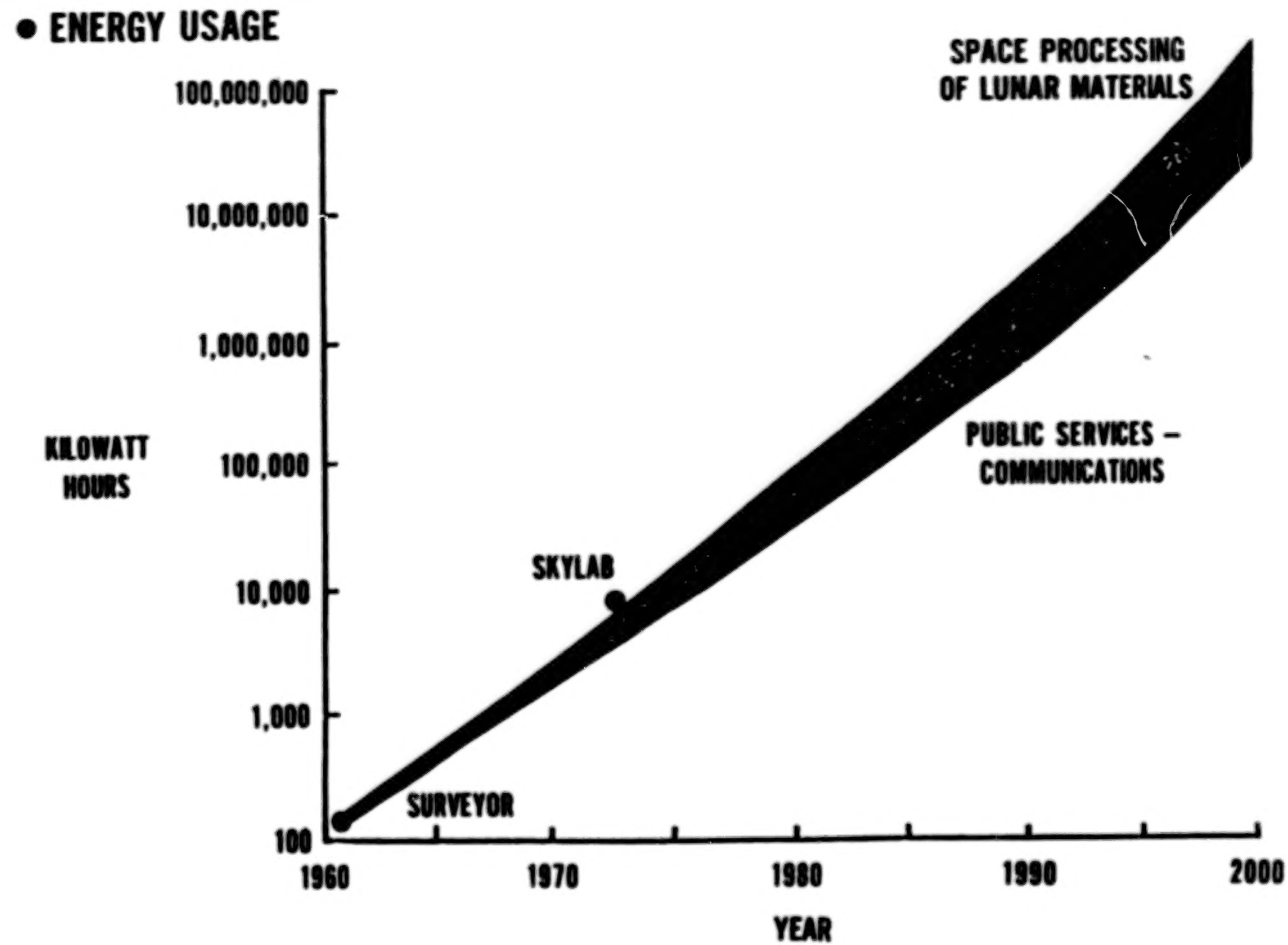


Figure 21

(Figure 22)

This figure depicts a mission concept, probably beyond the 1990s, representing the intensive phase of Space Exploration. We are looking at the surface of one of Jupiter's moons, with Jupiter in the upper left-hand corner. A nuclear electric powered, automated laboratory is in orbit. This laboratory is capable of interrogating, relaying, and analyzing information, perhaps even material collected from Jupiter and i' many moons. A JPL study (Automated Planetary Station at Jupiter) has identified, among other technology problems associated with this mission, the need for automated capability for operation of both the auxiliary equipment delivered to the planetary and satellite surfaces and the automated station itself.

Figure 22

65

(Figure 23)

This figure shows the historical experience and a projection of trends in automation. The measure shown here is the number of operations performed for each command sent to the spacecraft. It is interesting to note that NASA's so-called automated space program was never truly automated; the spacecrafts were really teleoperator systems in which each operation was directed by a single command. It was not until the Viking program that we made the first step toward an average of ten operations for one command. A typical command was the instruction "collect a sample and analyze it" in which the Viking scooped up a sample, fed it into a hopper, and samples were distributed among the different instrument packages for analysis. Future programs such as the multipurpose space services platform require increasing orders of magnitude growth in autonomous capability.

Economic demands also require us to move toward automated systems and they are the real drivers in our Global Services (or Applications) systems in earth orbit.

SPACE SYSTEM TECHNOLOGY FORECASTS – SPACECRAFT SYSTEMS

• AUTOMATION

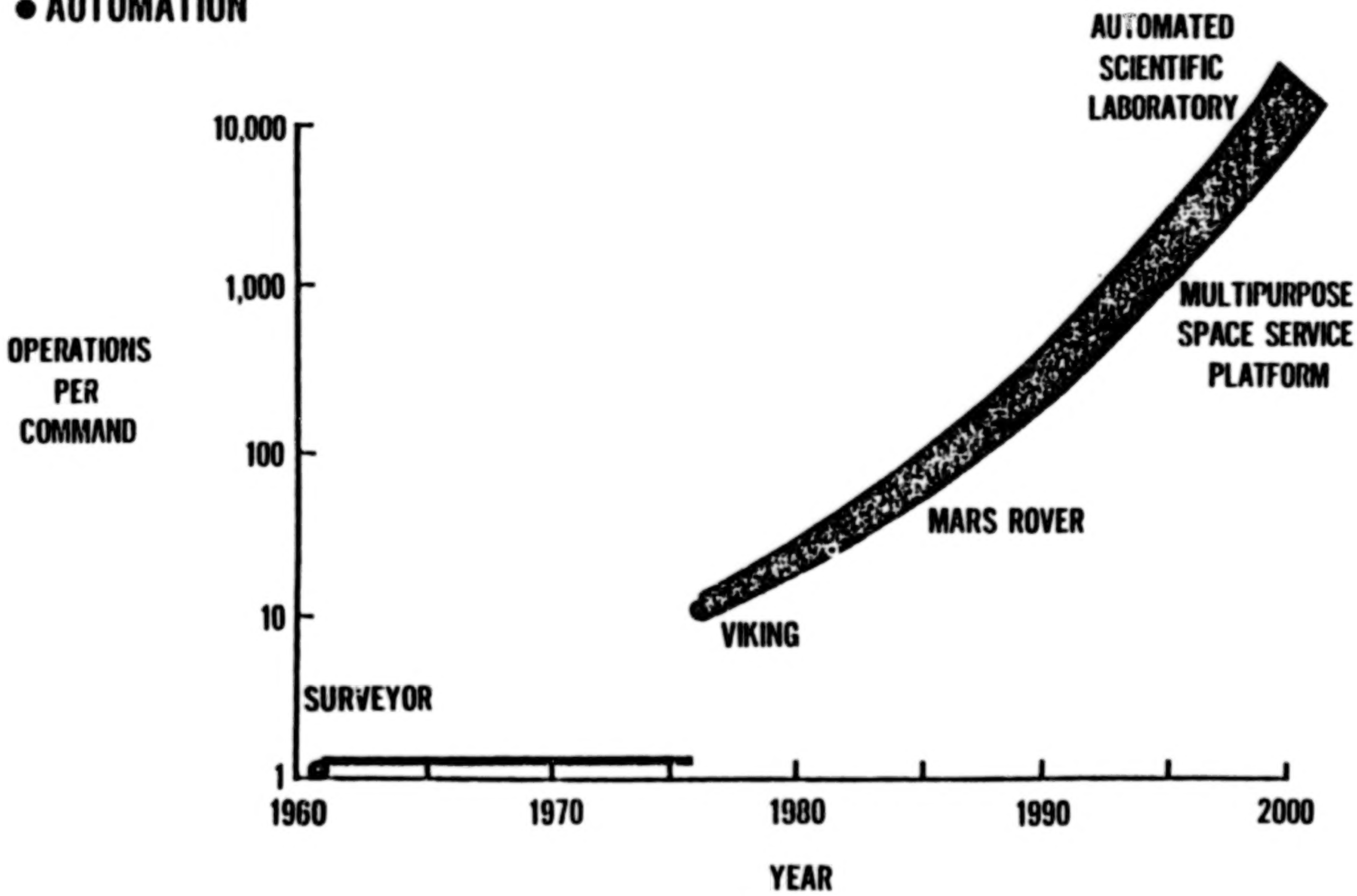
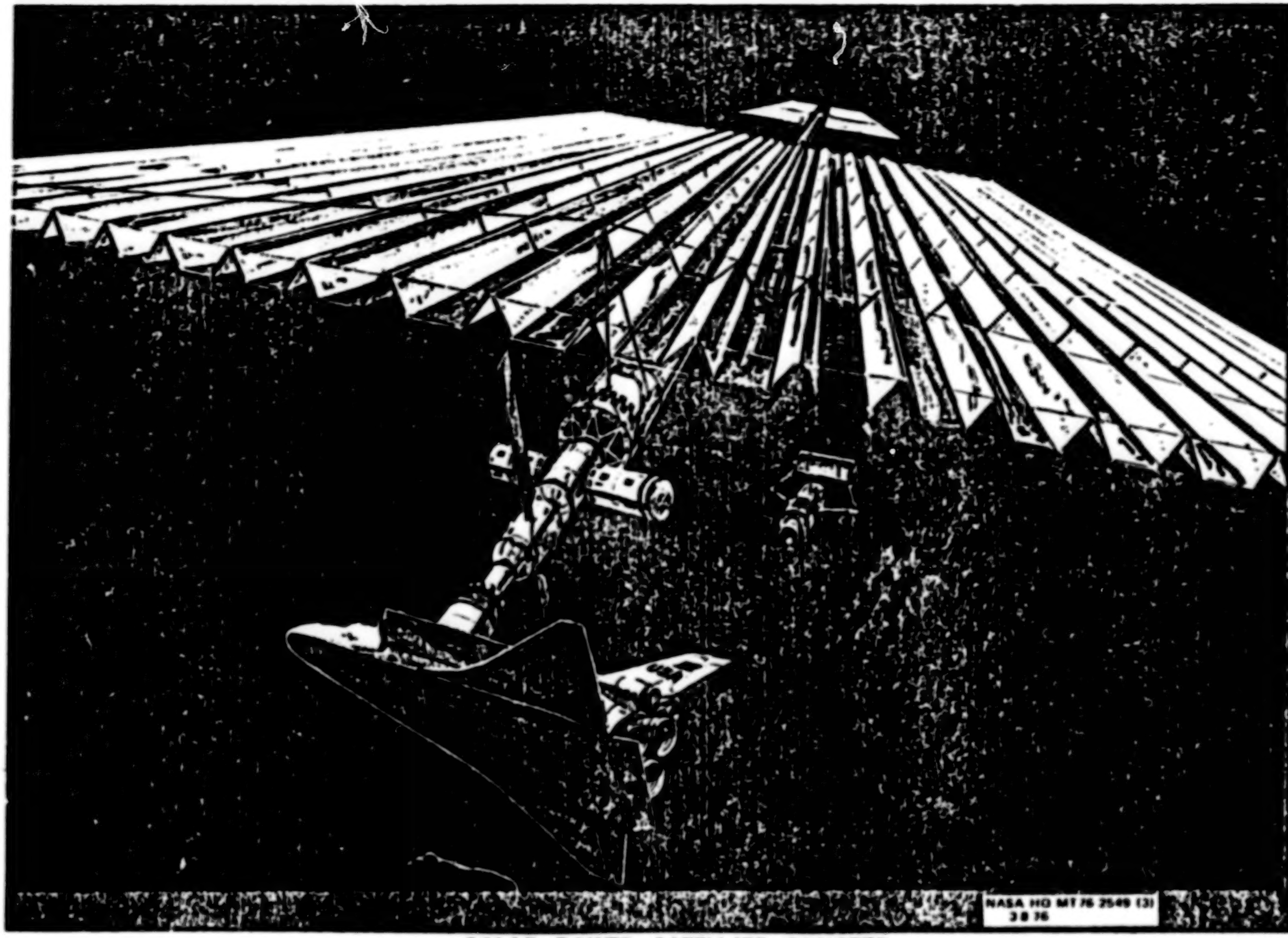


Figure 23

67

(Figure 24)

Now moving to perhaps the ultimate program for technology needs and opportunities, we come to an artist's rendition of that great glass palace in the sky--the solar power satellite (SPS). The SPS is a system probably for the turn of the century. We may certainly say that all of the aforementioned technology needs and opportunities converge on the SPS system. In fact, SPS stands in a unique driving position with respect to technology planning. Planning scenarios for the long-range are generally displayed both with and without SPS as a contributing element since its impact can be overwhelming.



SOLAR POWER SATELLITE CONCEPT

Figure 24

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SHUTTLE CREW STATION

ASTRONAUT INTERFACES

GEORGE C. FRANKLIN
NASA JOHNSON SPACE CENTER
JANUARY 17, 1978

71

72

(Figure 1)

The presentation that follows will describe the current orbiter configuration, the planned activities scheduled for early flight and a review of some of the facilities at JSC that are utilized to verify crew interfaces.

- BASIC ORBITER CREW INTERFACES
- RMS OPERATIONAL CAPABILITY
- FLIGHT CREW ACTIVITIES FOR EARLY FLIGHTS
- SUMMARY OF DEVELOPMENT FACILITIES AT JSC FOR CREW INTERFACES

Figure 1

73

74

(Figure 2; Figure 3)

The Orbiter has been designed to accommodate the work and off-duty activities of the crew. The mid-deck is the primary off-duty area, contains the airlock that allows access to the payload bay. The flight deck is the primary work area for all phases of the flight. Such a separation, within certain limits, will allow two shift operations for some missions.

The payload bay has crew provisions to support extravehicular activities by a suited crewman.

BASIC ORBITER CREW INTERFACES

- CREW MODULE

  DIVIDED INTO THREE PRINCIPAL ELEMENTS ACCESSIBLE BY THE CREW:

  - MID-DECK - PRIMARY OFF-DUTY AREA
  - AIRLOCK - EVA ACCESS TO PAYLOAD BAY
  - FLIGHT DECK - PRIMARY WORK AREA

- PAYLOAD BAY

  PROVISIONS TO SUPPORT EXTRAVEHICULAR ACTIVITIES (EVA).

Figure 2

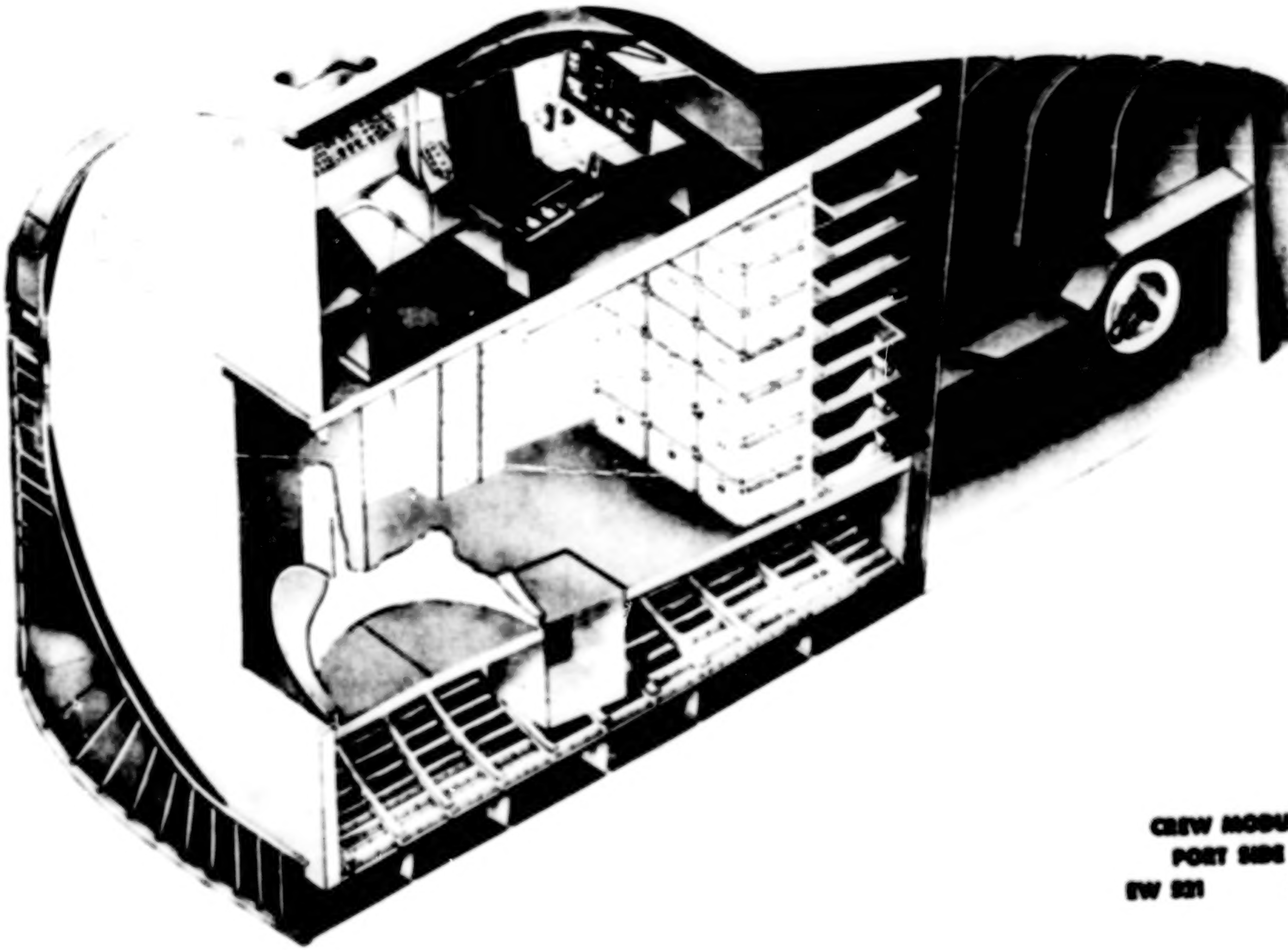


Figure 3

78

(Figure 4 Figure 5 Figure 6)

Chart 5 shows the arrangement of the stowage lockers on the forward avionics bay of the mid-deck. It indicates the sleeping bunks on the starboard side. The operational bunks are enclosed with individual ventilation and light control.

Chart 6 shows the galley on the port side and the waste management-hygiene provision to the aft port side. The airlock is located on the aft bulkhead, centered on the mid-deck. Some additional lockers are available to the starboard side of the airlock against the aft avionic bay.

<u>CREW MODULE PROVISIONS</u>

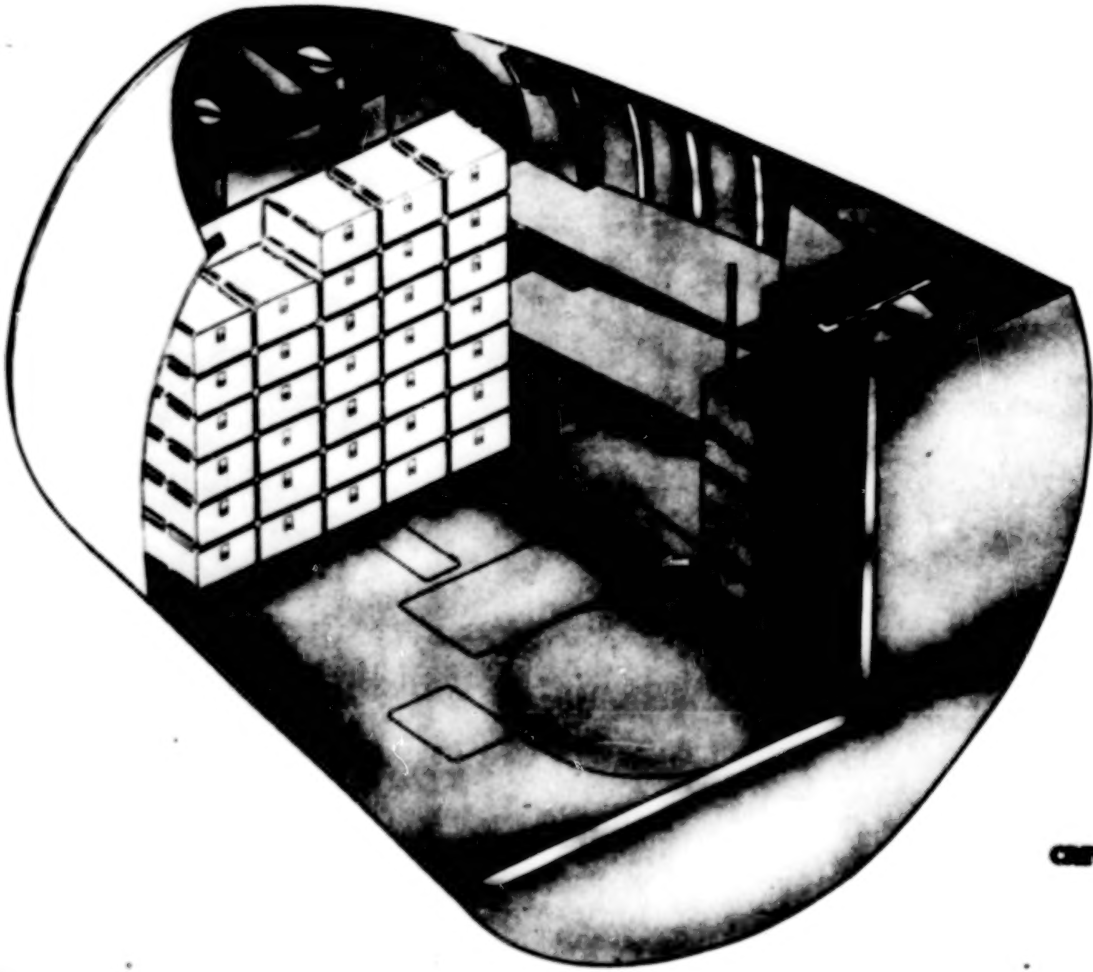
- MID-DECK PROVIDES THE OFF-DUTY PROVISIONING AND FACILITIES FOR THE CREW
    - STOWAGE PROVISION
        - EQUIPMENT FOR CREW COMFORT
        - MISSION UNIQUE EQUIPMENT
    - SLEEPING ACCOMMODATIONS
    - FOOD PREPARATION AND EATING PROVISIONS
    - HYGIENE PROVISIONS
    - WASTE MANAGEMENT

- AIRLOCK - PROVIDES EVA ACCESS TO THE PAYLOAD BAY
    - INCLUDES EVA EQUIPMENT STOWAGE
    - MAY BE EITHER INSIDE MODULE OR IN PAYLOAD BAY DEPENDING ON MISSION NEEDS

Figure 4

NASA
S- 75- 31238

80



CREW MODULE, MID-DECK
STARBOARD

Figure 5

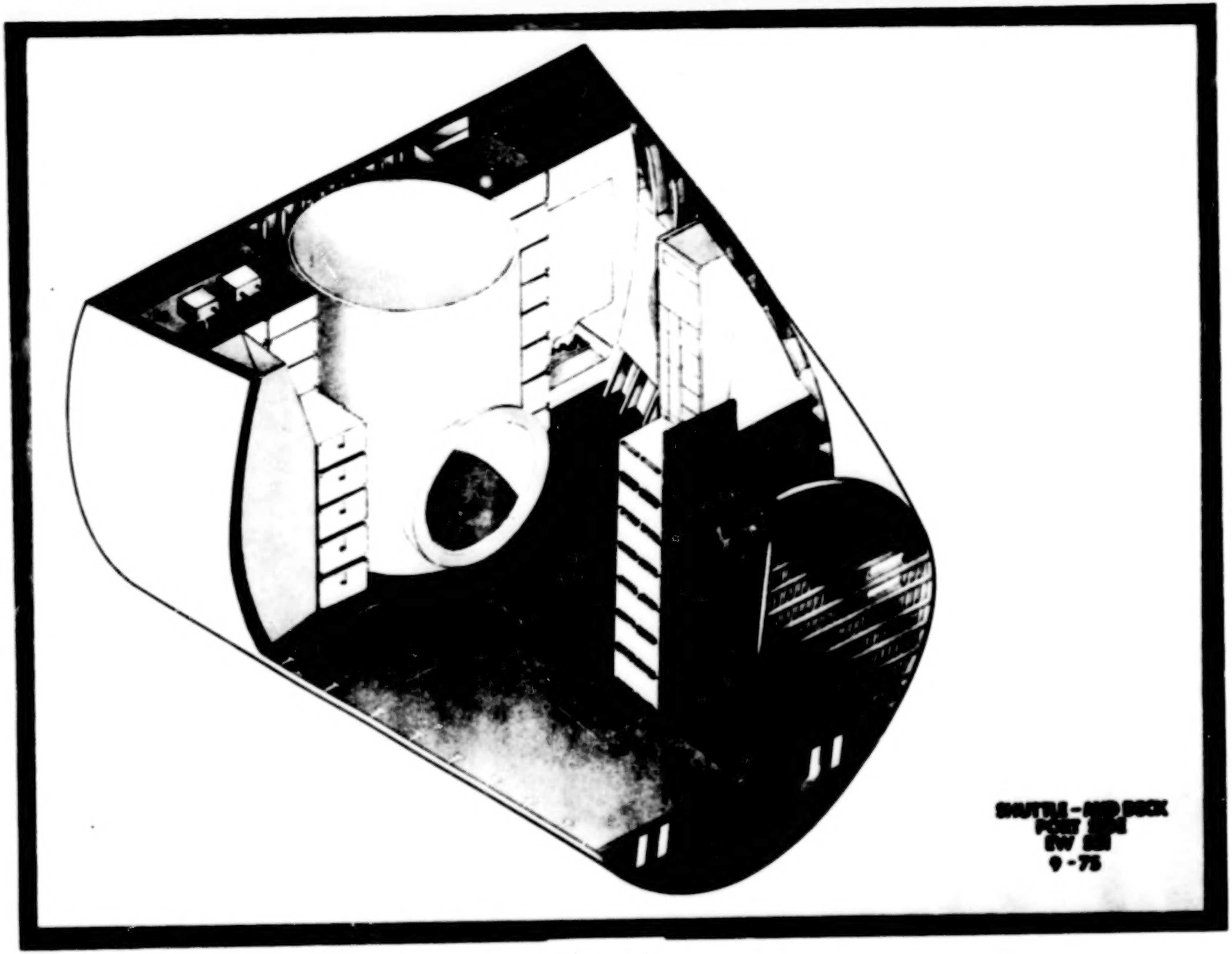


Figure 6

81

82

(Figure 7·Figure 8;Figure 9)

Chart 8 - The forward flight deck is the control station for the pilot and commander during boost, reentry and landing. It may also be utilized on orbit as required. The aft flight deck contains the mission specialist station on the starboard side. Critical payload functions during launch and reentry can be controlled from this station. The orbiter vehicle systems are also controlled from this area.

Chart 9 - Payload specialist is located on the port side of the aft flight deck. The orbit station for rendezvous, RMS operations and other operations requiring control of payload systems with exterior visibility are carried out utilizing the aft facing consoles.

CREW MODULE PROVISIONS

(CONTINUED)

- FORWARD FLIGHT DECK - CONTROL STATION FOR THE COMMANDER AND PILOT DURING BOOST, REENTRY AND LANDING

- AFT FLIGHT DECK
    - MISSION SPECIALIST - CONTROL OF VEHICLE SYSTEMS
    - PAYLOAD SPECIALIST - CONTROL OF PAYLOAD ACTIVITIES INDEPENDENT OF ORBITER SYSTEMS
    - ON ORBIT STATION FOR RENDEZVOUS, RMS OPERATIONS, AND OTHER OPERATIONS REQUIRING EXTERIOR VISIBILITY TO THE PAYLOAD BAY OR OVERHEAD

Figure 7

NASA
S-75-31239

84

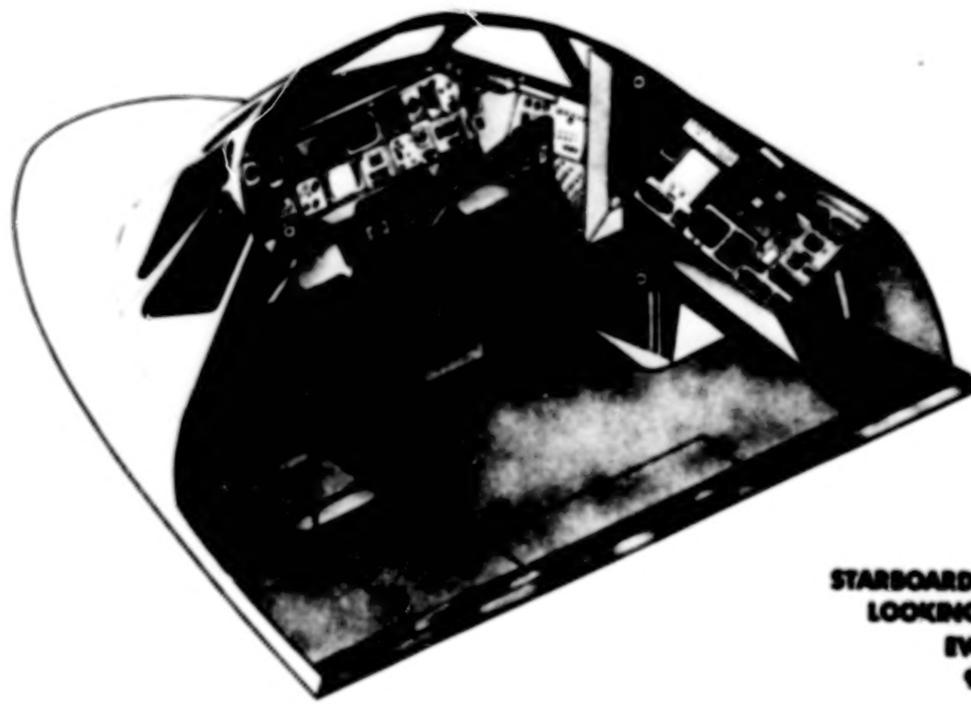


Figure 8

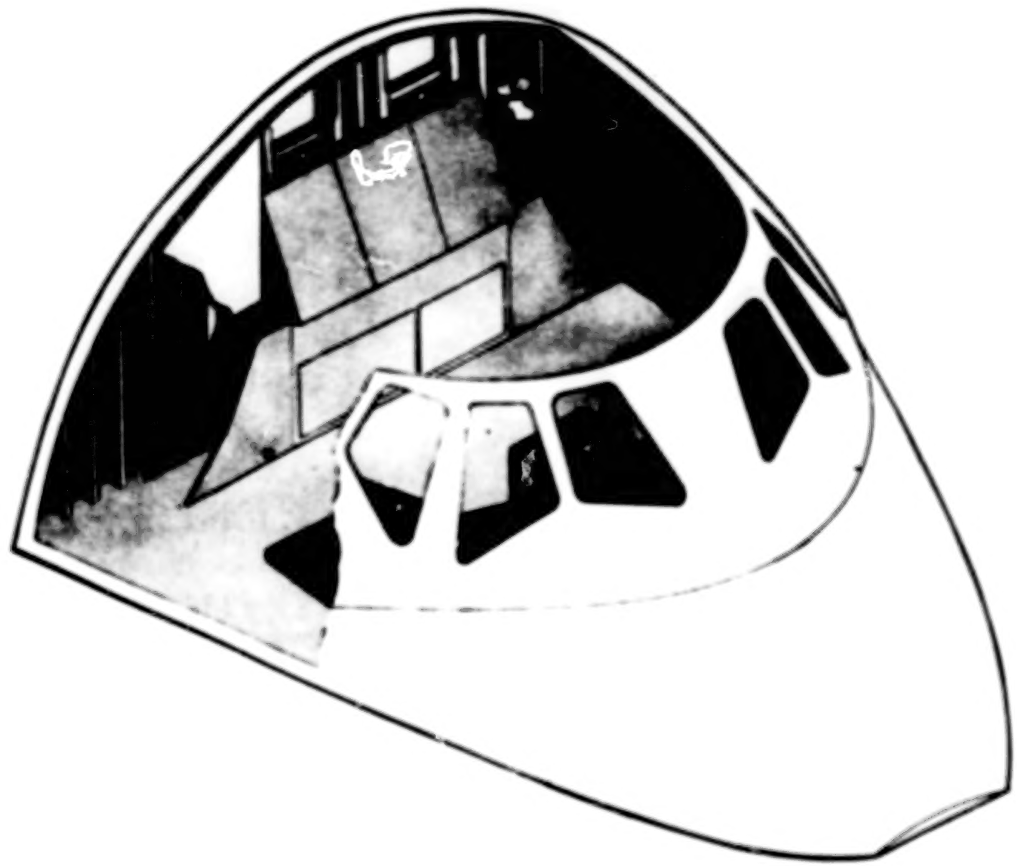


Figure 9

CONTENTS

(Figure 10 Figure 11 Figure 12, Figure 13)

The aft flight deck principal features are shown.

There is adequate room for two crewmen operating the aft consoles and in some mission phases, additional crewmen can operate in the aft flight deck. Most functions carried on have been identified as sequential, thus not requiring more than two operators simultaneously.

MISSION SPECIALISTS

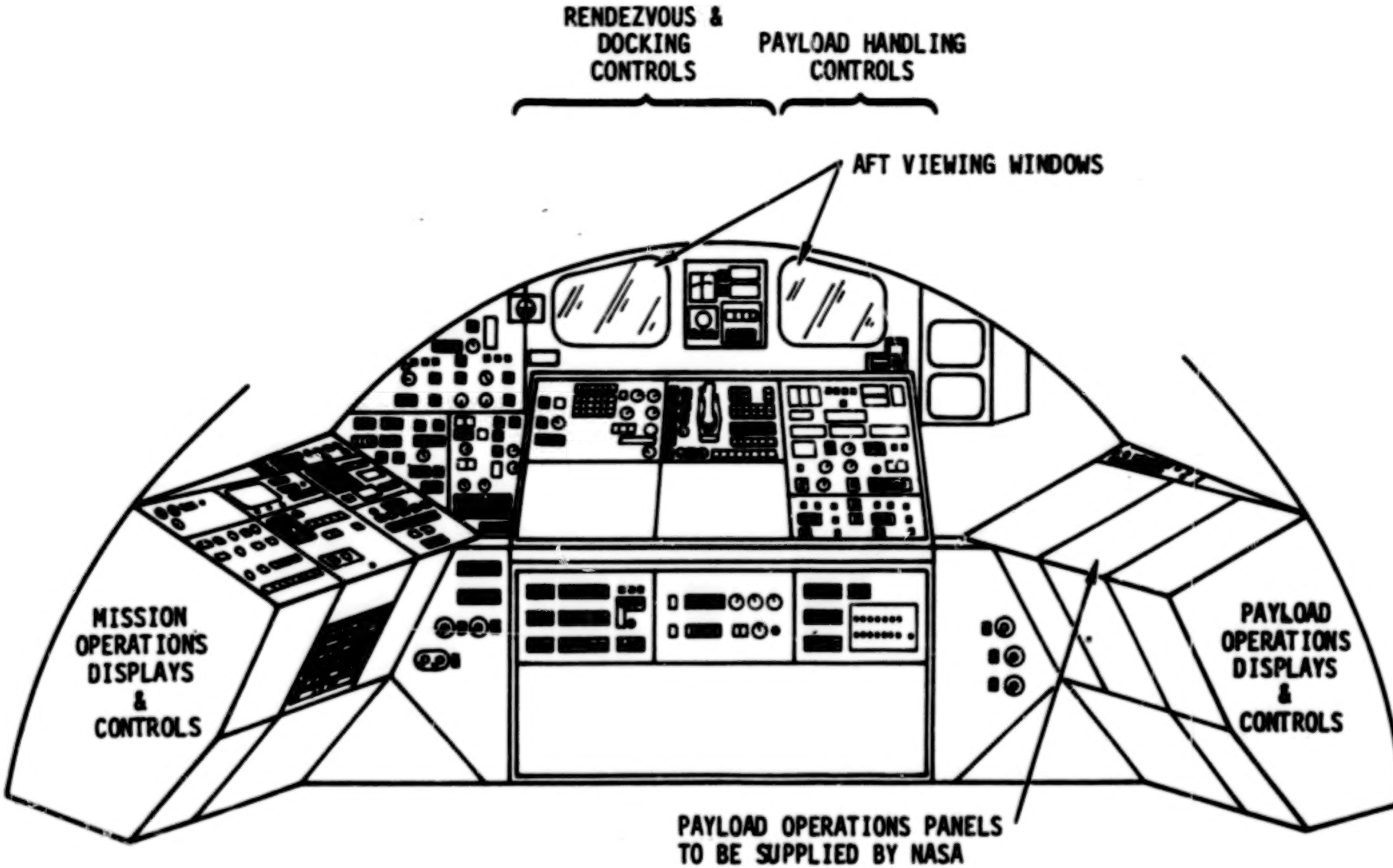
- COMMUNICATION SYSTEM CONTROL
  - INTERCOM CONFIGURATION
  - UPLINK/DOWNLINK VOICE, TV, AND DATA
- POWER MANAGEMENT AND SYSTEM CONTROL
- OTHER VEHICLE SYSTEMS MANAGEMENT AND CONTROL
- CONFIGURED WITH CRT/KEYBOARD FOR SYSTEM FLEXIBILITY AND SPECIFIC CONTROLS AND DISPLAYS WHERE OPERATIONS DICTATE

Figure 10

87

88

AFT FLIGHT DECK CONFIGURATION



VIEW LOOKING AFT

Figure 11

PAYLOAD SPECIALISTS

- CURRENT PLANNING INTENDS TO PROVIDE A STANDARD SET OF DISPLAYS AND CONTROLS FOR PAYLOAD USERS.
    - CRT/KEYBOARD
    - AN ARRAY OF SWITCHES WITH STATUS DISPLAYS
    - POINTING CONTROLS
- SPACE WILL ALSO BE AVAILABLE FOR MISSION UNIQUE DISPLAYS AND CONTROLS WHERE REQUIRED.
- ADDITIONAL AREA IS PROVIDED AT THE MISSION SPECIALIST AND ON ORBIT CONSOLES FOR PAYLOAD UNIQUE CONTROLS WHEN REQUIRED.

Figure 12

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ON ORBIT STATION

- REMOTE MANIPULATOR SYSTEM (RMS) CONTROLS AND DISPLAYS.
- TWO TV MONITORS WITH SPLIT SCREEN
- TV CONTROLS - CAMERA/MONITOR SELECTION, TILT, PAN AND ZOOM FEATURES PROVIDED
- PAYLOAD BAY LIGHTING
- TERMINAL RENDEZVOUS, STATION KEEPING
- OVERHEAD AND PAYLOAD BAY DIRECT VISION CAPABILITY
    - SUNSHADES AND GLARE FILTERS ARE PROVIDED
- FLIGHT DECK STOWAGE PROVIDED FOR DATA FILES, CHECK LISTS, ETC.

Figure 13

92

(Figure 14)

The principal documents that describe the vehicle interfaces are indicated. The ICD's are expected to replace Volume XIV as the program matures.

There are many options that cannot be put into an ICD, therefore questions should be asked of the STS office or the Crew Station Branch personnel when required.

AFT FLIGHT DECK

PRINCIPAL REFERENCE DOCUMENTS

ICD'S

- "SHUTTLE ORBITER/CARGO STANDARD INTERFACES" 2-19001
- "INTEGRATED AFT FLIGHT DECK" 3-0014-01
- LEVEL II PROGRAM DEFINITION AND REQUIREMENTS, JSC-07700, VOLUME XIV - SPACE SHUTTLE SYSTEM PAYLOAD ACCOMMODATIONS

Figure 14

94

(Figure 15 Figure 16 Figure 17)

The design eye shown on charts 16 and 17 should only be utilized for reference. The crewman is free to move his head anywhere to gain a better view out the window. We have noted after exercises on the Manipulator Development Facility window smudges caused by the operator pressing his/her nose against the glass in the endeavor to gain better vision into the payload bay. There are no significant "blind" spots between the overhead and aft facing window when the crewman moves his/her head.

AFT FLIGHT DECK VISIBILITY

- DESIGN EYE SHOULD BE UTILIZED FOR REFERENCE PURPOSES.

- CREWMAN HAS FREEDOM OF MOVEMENT AT THE AFT STATION THUS IMPROVING VISIBILITY INTO THE PAYLOAD BAY AND OVERHEAD.

Figure 15

96

PORT REAR VIEW WINDOW
VISION ANGLES AT DESIGN EYE POSITION

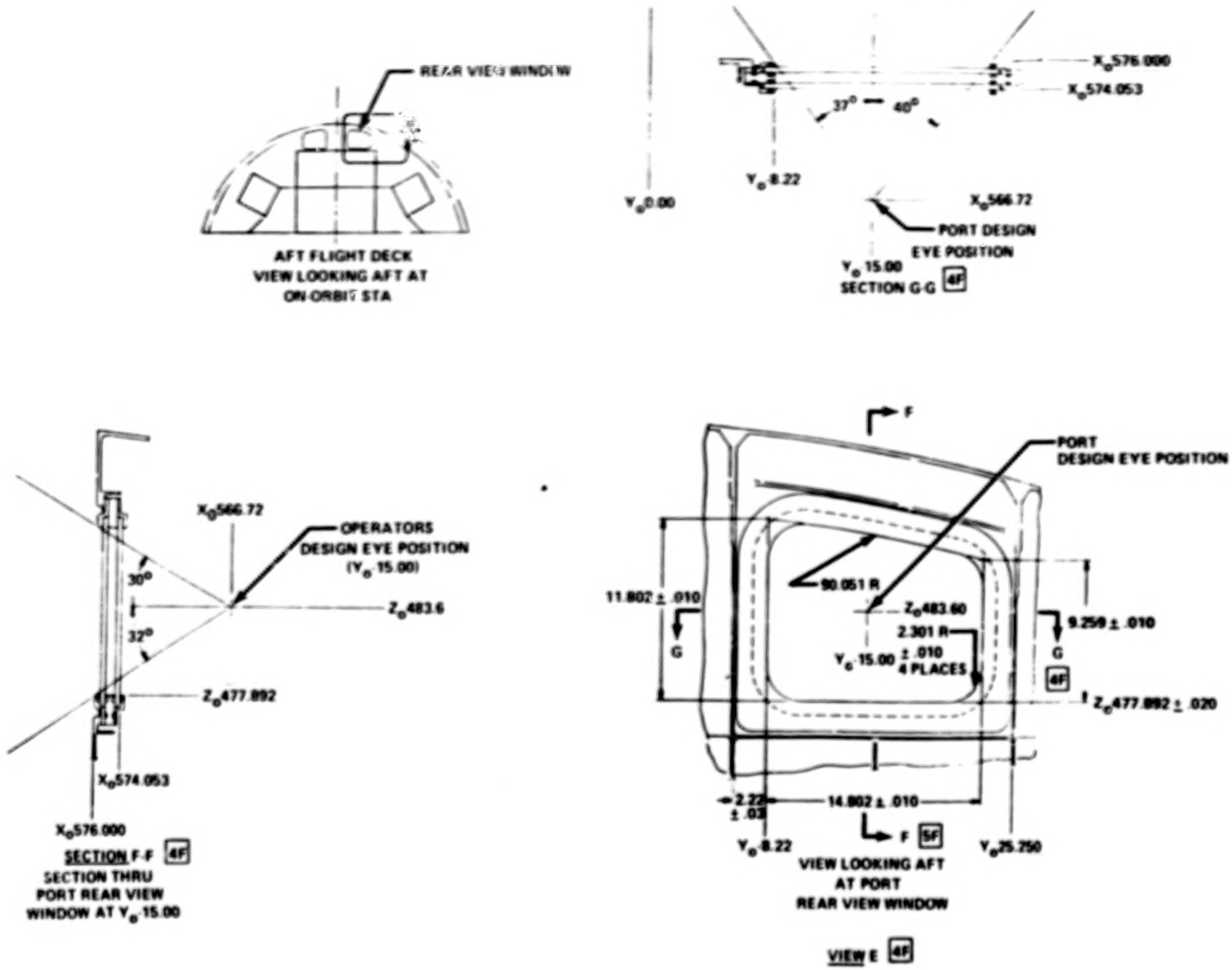

Figure 16

PORT OVERHEAD WINDOW VISION ANGLES AT DESIGN EYE POSITION

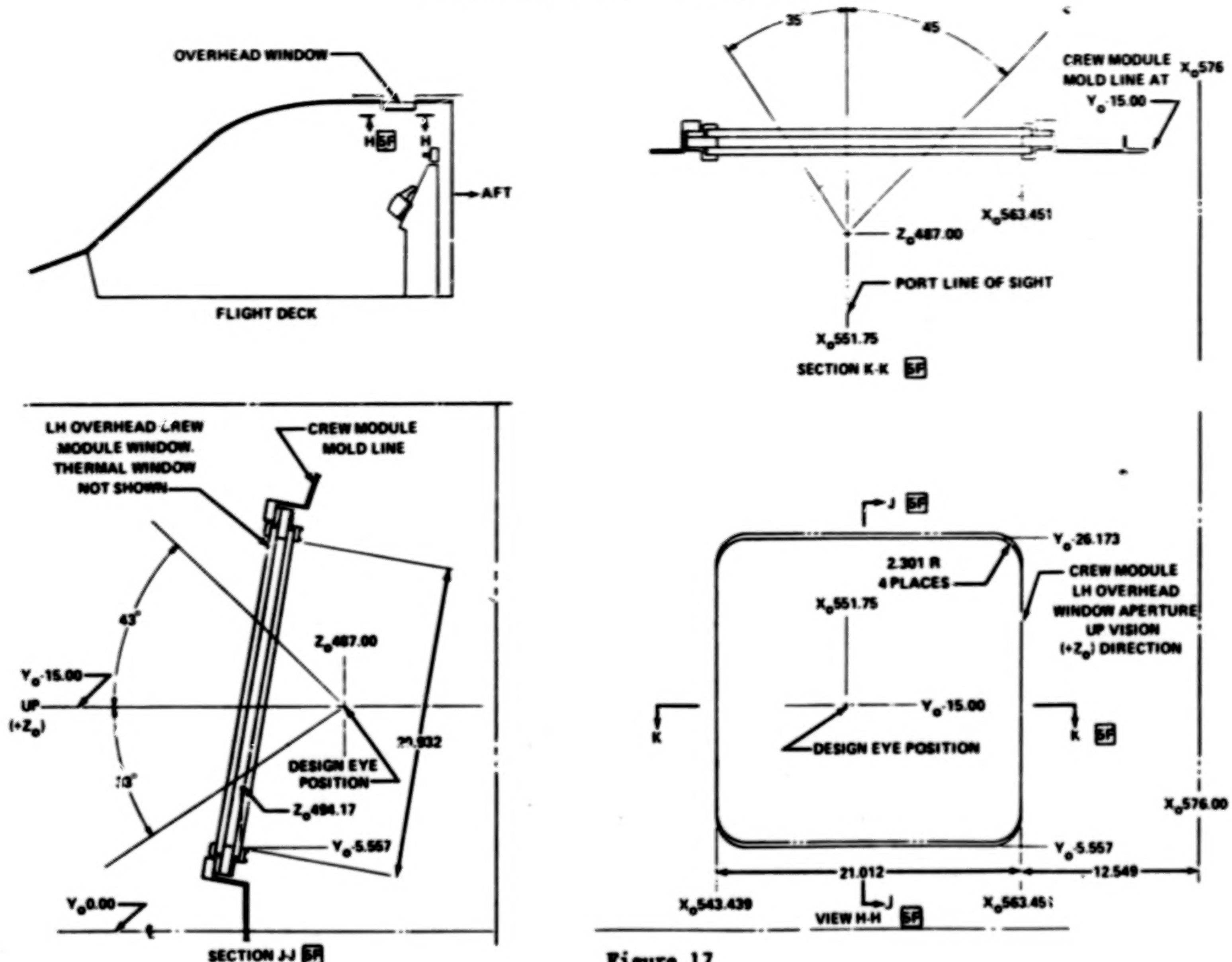

Figure 17

(Figure 18; Figure 19)

The orbiter provides hand holds as mobility aids to the EVA crewman, which will allow him to traverse down either door hingeline and across each of the end bulkheads. These hand holds also provide tether and portable work station attachment capability for the EVA crewman.

PAYLOAD BAY EVA INTERFACES

- STANDARDIZED HAND HOLDS ARE PROVIDED ON THE FORE AND AFT BULKHEADS AND ON BOTH SIDES OF THE DOOR HINGE LINE.
  - PROVIDES MOBILITY PATH FROM AIRLOCK HATCH TO ALL PARTS OF THE PAYLOAD BAY FOR THE EVA CREWMAN.
  - PROVIDES ATTACHMENT CAPABILITY FOR THE FURNISHED PORTABLE WORK STATION.

Figure 18

100

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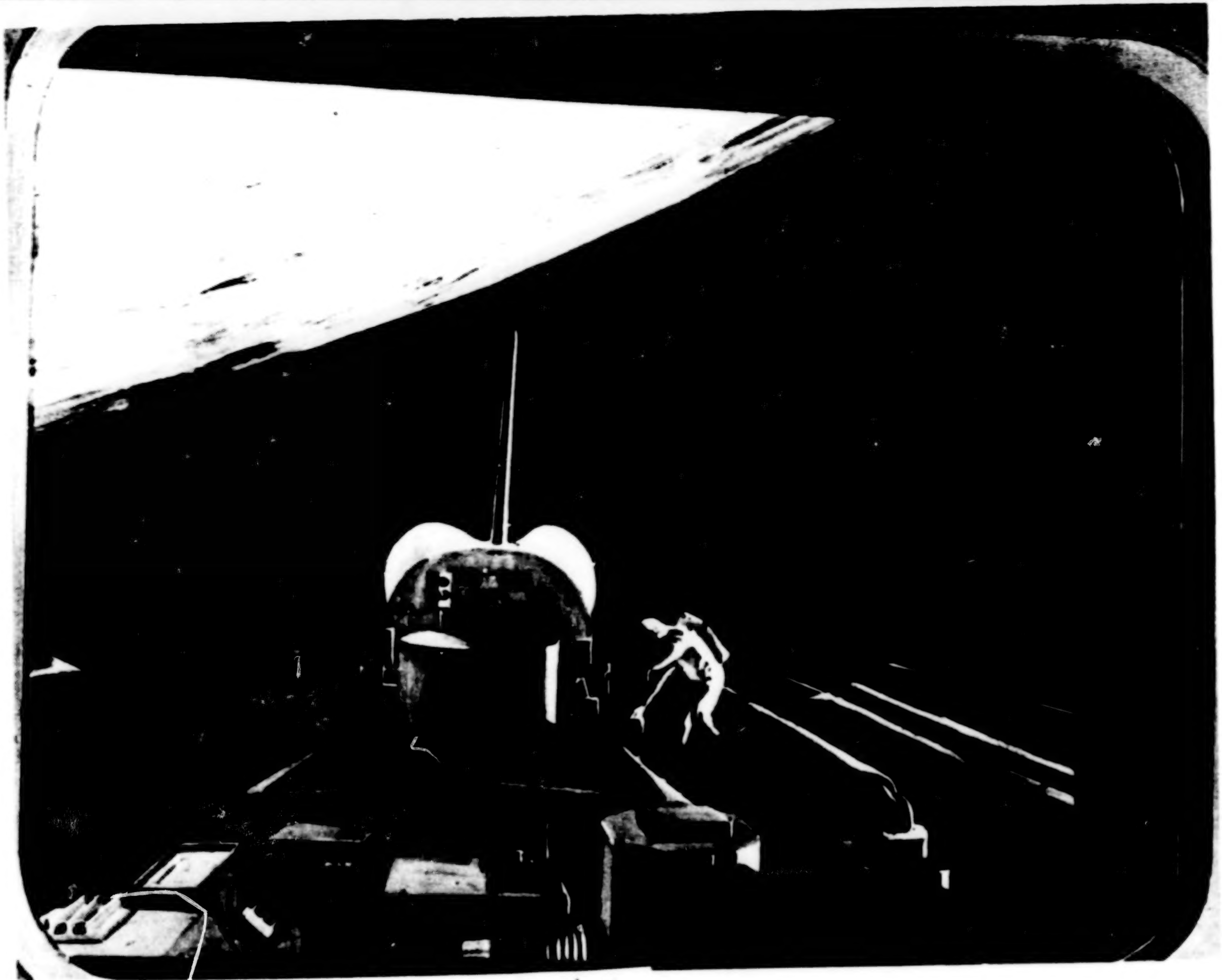

Figure 19

102

(Figure 20; Figure 21)

The portable work station is planned as a piece of standard orbiter hardware. If payloads have interfaces similar to the hand holds or bridge fitting interfaces, it could be utilized. The foot restraint is an upgraded Skylab design that allows the EVA astronaut to keep both hands free to work.

PORTABLE EVA WORK STATION

- A PORTABLE EVA WORK STATION IS AVAILABLE TO SUPPORT PAYLOAD RELATED ACTIVITIES.
  - DESIGNED TO ATTACH TO HAND HOLDS AND BRIDGE FITTING INTERFACES.
  - PROVIDES CREWMAN RESTRAINT (FOOT) TO ALLOW TWO HANDED ACTIVITIES.
  - PROVIDES TEMPORARY STOWAGE OF EVA TOOLS.

Figure 20

103

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Figure 21

105

(Figure 22 ; Figure 23, Figure 24, Figure 25)

The data presented on charts 23, 24, and 25 describe the principal elements of the RMS. Anyone desiring details should contact the Payload Deployment and Retrieval Systems Office at JSC. The RMS is controlled by two hand controllers located at the aft facing consoles. The feedback is visual by gauges, TV or out-the-window direct vision. The operator can select the reference system best suited to the task, depending on where the payload-RMS combination may be.

REMOTE MANIPULATOR SYSTEM (RMS)

- CONTROLLED BY TWO 3 DEGREE-OF-FREEDOM (DOF) HAND CONTROLLERS
    - TRANSLATION CONTROLLER
    - ROTATION CONTROLLER
- HAS BOTH AN AUTO AND MANUAL MODE OF CONTROL, SELECTABLE BY THE OPERATOR
- HAS FOUR COORDINATE REFERENCE SYSTEMS SELECTABLE BY THE OPERATOR
    - ORBITER
    - END EFFECTOR
    - PAYLOAD
    - PAYLOAD/ORBITER

Figure 22

ELEMENTS OF THE RMS INSTALLATION

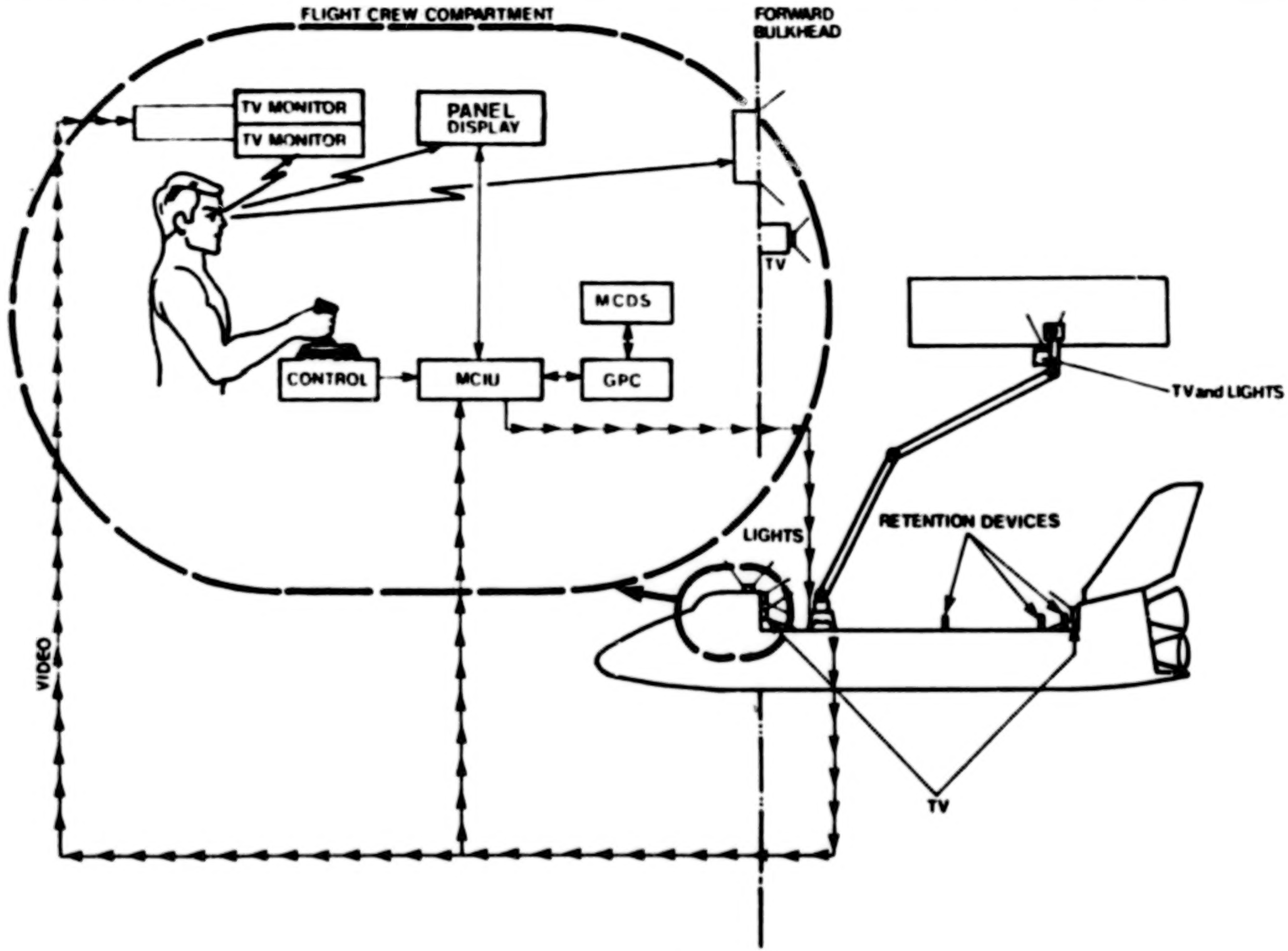

Figure 23

RMS BASELINE

JOINT D.O.F.	GUARANTEED TORQUE (LB. FT.)	OUTPUT SPEED		ANGULAR MOVEMENT DEGREES
		LOADED SPEED (RAD/SEC)	UNLOADED SPEED (RAD/SEC)	
SHOULDER				
- YAW	772	0.0040	0.040	±180
- PITCH	772	0.0040	0.040	0 TO -145
ELBOW				
- PITCH	528	0.0056	0.056	0 TO +160
WRIST				
- PITCH	231	0.0083	0.083	±120
- YAW	231	0.0083	0.083	±120
- ROLL	231	0.0083	0.083	0 ±450

MAX REACH - 50'
TIP FORCE - 15 LBS +
TIP SPEED - VARIABLE -2.0'/S MAX

Figure 24

110

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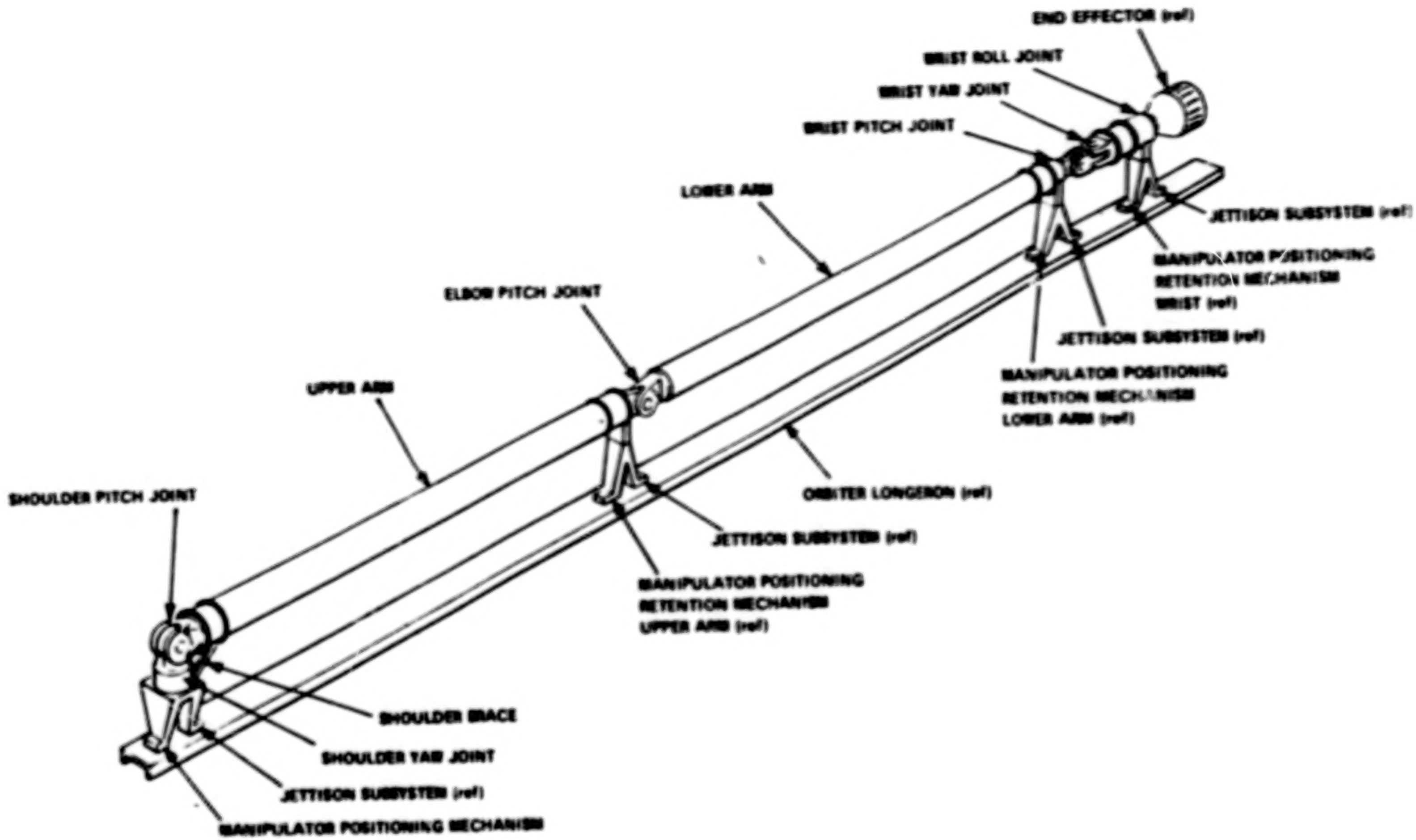

MECHANICAL ARM ASSEMBLY

Figure 25

(Figure 26)

The flight program is divided into two principal groups - orbital flight test (OFT) and operations/Shuttle Transportation System (STS).

FLIGHT CREW ACTIVITIES

- ORBITAL FLIGHT TEST (OFT)

- OPERATIONAL FLIGHTS/SHUTTLE TRANSPORTATION SYSTEM (STS)

Figure 26

114

(Figure 27)

The OFT program is utilized to test the orbiter capability and prepare it for operational status. Two crewmen are utilized in the early flights. Payload activities are kept to a minimum. We do have an option to fly four crewmen on the last OFT flights. The Skylab Boost mission has been recently established earlier than this chart indicates.

ORBITAL FLIGHT TEST SUMMARY

- PRINCIPAL OBJECTIVES OF OFT IS TO VERIFY ORBITER CAPABILITY.
- TWO CREWMEN TO OPERATE FLIGHTS 1-4
    - LIMITED AFT FLIGHT STATION ACTIVITY TO OPERATE SMALL PALLET AND INSTRUMENTS
    - MANIPULATOR CHECKED OUT FLIGHT #3
    - DEPLOYMENT OF ACTIVE PAYLOAD FLIGHT #4
- CAPABILITY TO FLY 4 CREWMEN ON OFT FLIGHTS 5 AND 6
    - SKYLAB REVISIT/BOOST MISSION TENTATIVELY PLANNED
    - UPPER STAGES TO BE DEPLOYED
- CONTINGENCY EVA CAPABILITY IS PROVIDED FOR EACH FLIGHT.
- A FLIGHT TEST OBJECTIVE (FTO) IS BEING REVIEWED FOR FLIGHT #6 TO CONDUCT A SCHEDULED EVA (DECISION IS PENDING).

115

Figure 27

(Figure 28; Figure 29, Figure 30, Figure 31, Figure 32)

The subsequent charts indicate the STS flight assignment baseline that is controlled by the STS program. Beginning with flight 7, the missions are divided as indicated into short delivery-retrieval missions or longer duration Spacelab missions. There is no planned EVA in the program at this time, however, some of the payload designers are inquiring on the merits of using EVA in lieu of automating or backing up some systems. As each of these missions mature, the crew activities become better known. We have been processing many requests for crew interfaces and their interaction with the payload handling or operation.

OPERATIONAL FLIGHTS

- VEHICLE CONSIDERED OPERATIONAL FROM FLIGHT 7 AND SUBSEQUENT.
- BASELINE SPACE TRANSPORTATION SYSTEM (STS) FLIGHT ASSIGNMENTS HAVE BEEN ESTABLISHED.
- DELIVERY MISSIONS VARY FROM 2 DAYS TO 5 DAYS DURATION WITH 3 CREWMEN.
- SPACELAB MISSIONS DURATION RANGE FROM 7 DAYS TO 12 DAYS WITH 5 CREWMEN ON BOARD.
- CONTINGENCY EVA CAPABILITY FOR EACH FLIGHT IS PLANNED, HOWEVER, NO PLANNED EVA IS SCHEDULED AT THIS TIME.

Figure 28

118

STS FLIGHT ASSIGNMENT BASELINE

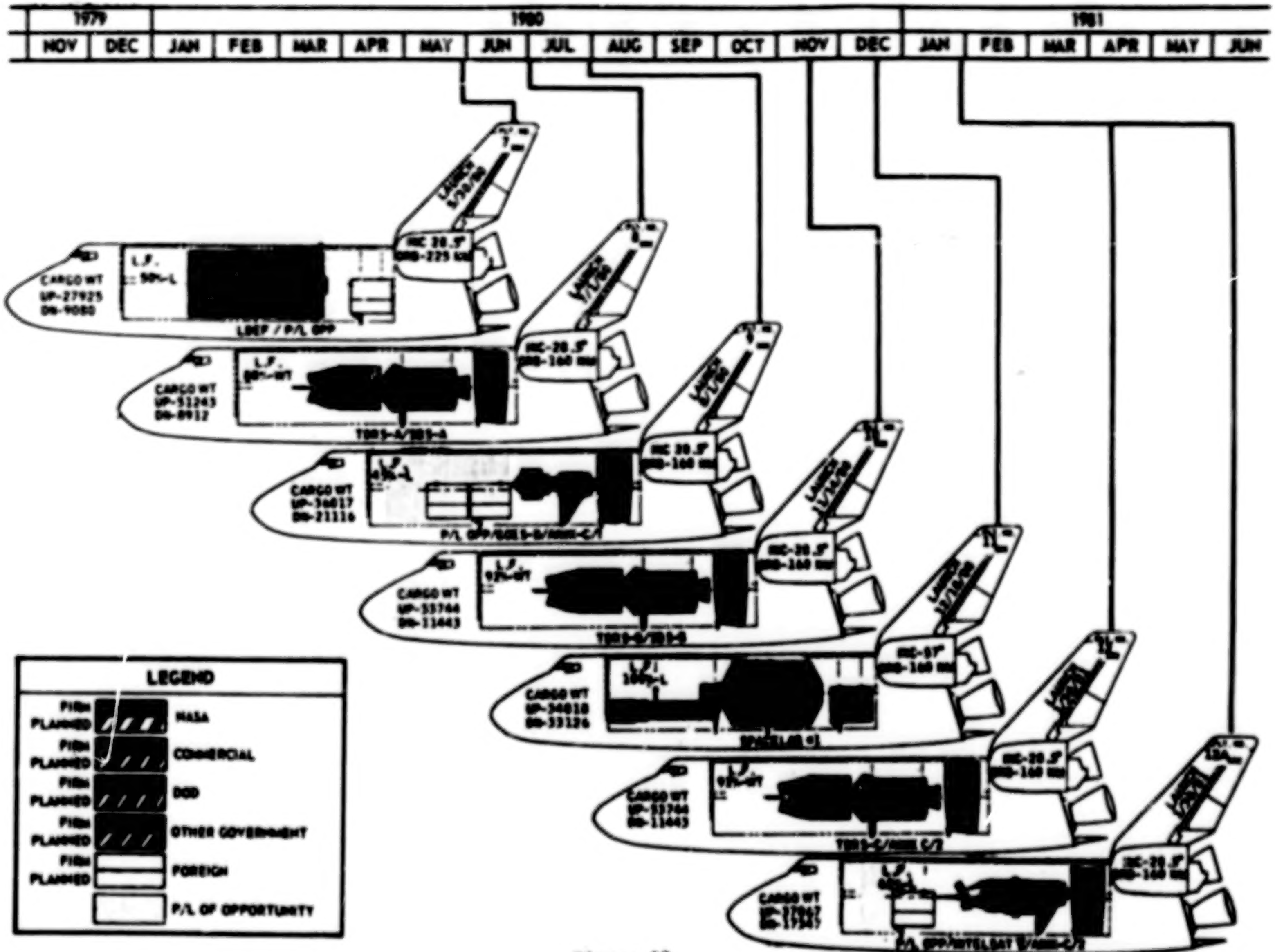

Figure 29

STS FLIGHT ASSIGNMENT
BASELINE

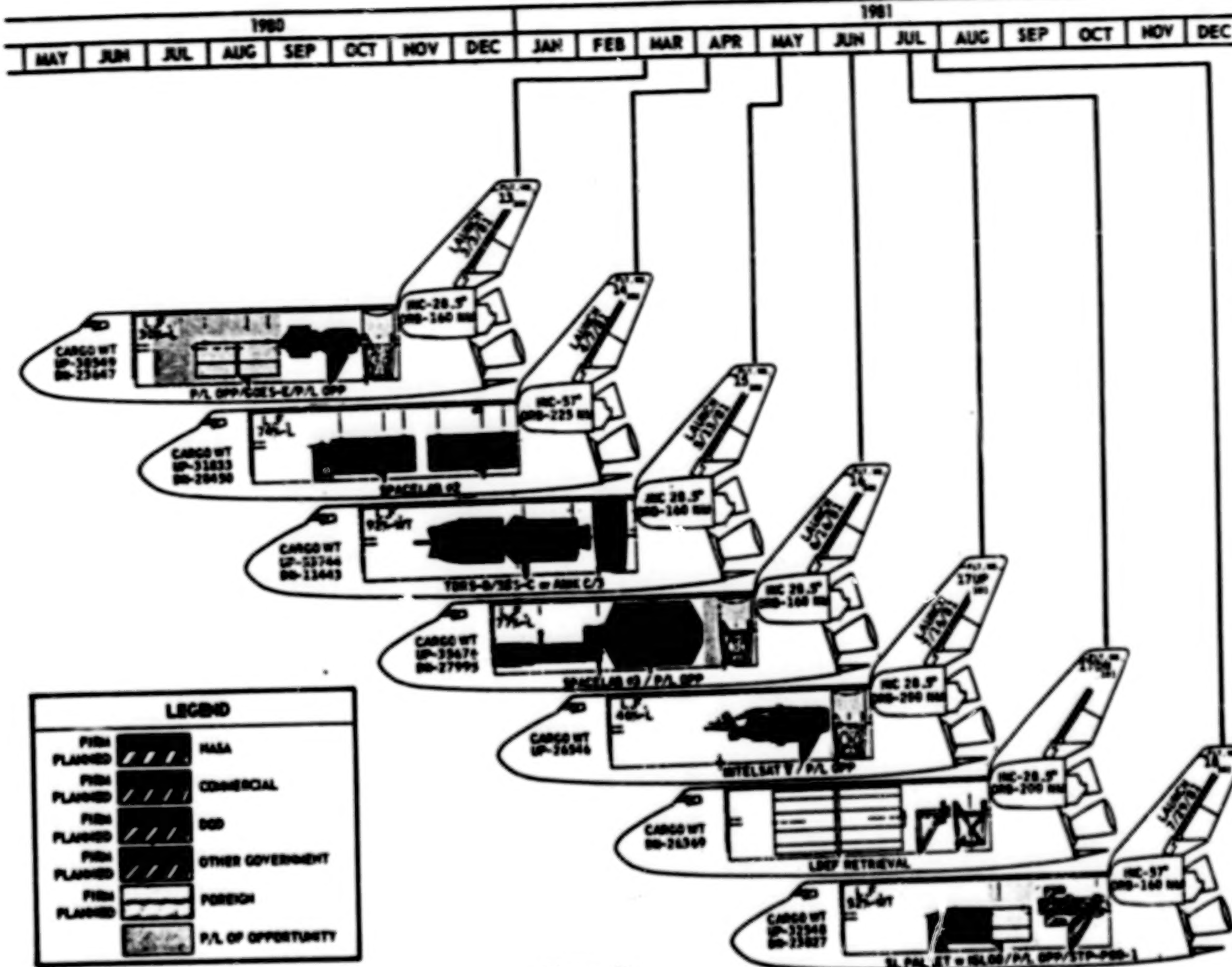

Figure 30

119

120

STS FLIGHT ASSIGNMENT
BASELINE

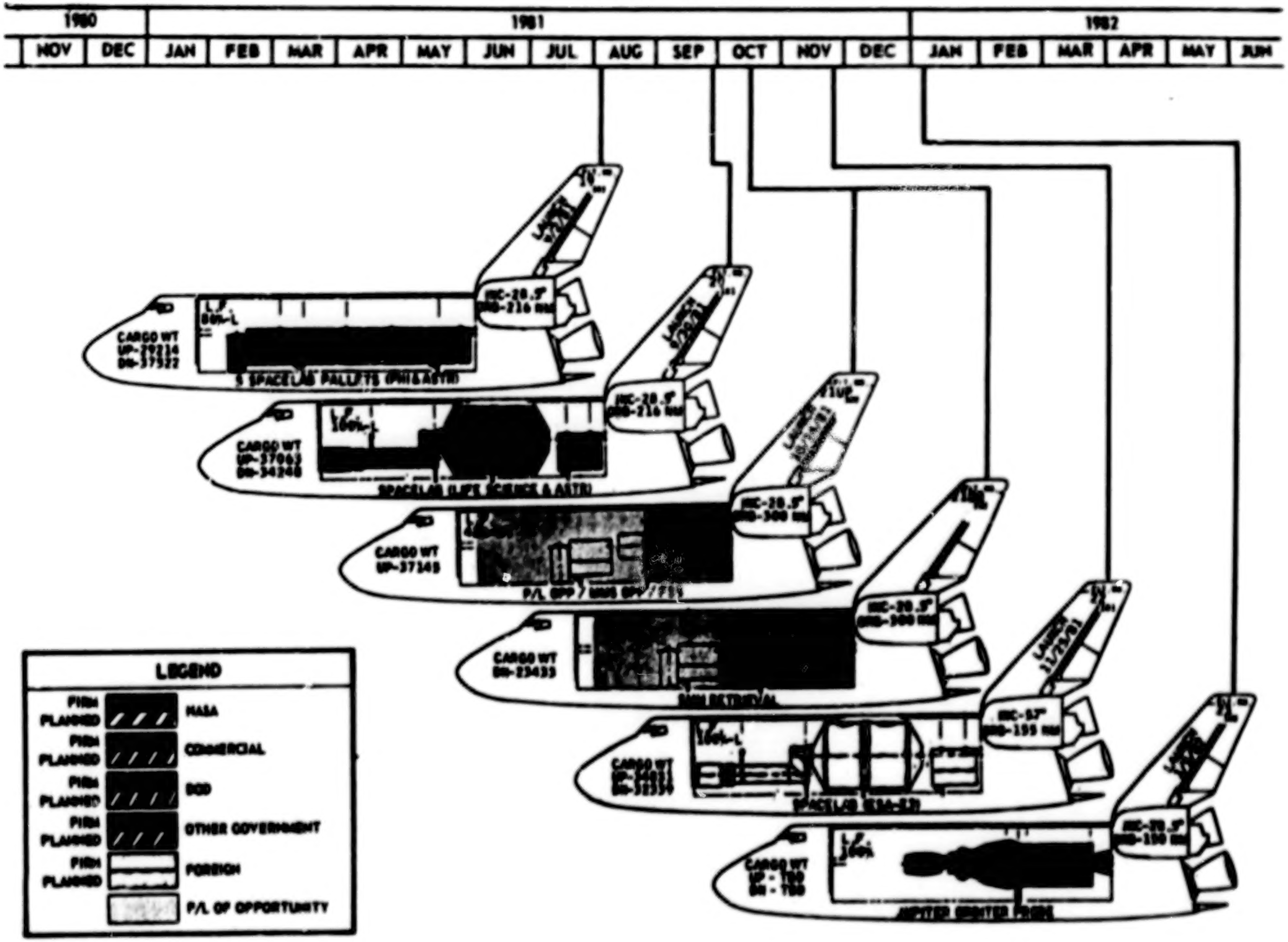

Figure 30

FLIGHT ASSIGNMENT BASELINE

FLIGHT NO.	PRELIMINARY LAUNCH DATE	CARGO †
7	5/30/80	LDEF DELIVER, (OFT PALLET OF OPPORTUNITY)
8	7/1/80	TDRS-A, SBS-A
9	8/1/80	(TWO PALLETS OF OPPORTUNITY), GOES-D, ANIK -C/1
10	11/14/80	TDRS-B, SBS-B
11	12/18/80	SPACELAB #1, LONG MODULE W/PALLET
12	1/30/81	TDRS-C/ANIK - C/2
12 ALTERNATE	1/30/81	(ONE PALLET OF OPPORTUNITY),* INTELSAT V, ANIK - C/2
13	3/3/81	(TWO PALLETS OF OPPORTUNITY), GOES-E,(SSUS-D OF OPPORTUNITY)
14	4/7/81	SPACELAB #2, 4 PALLETS W/IGLOO
15	5/13/81	TDRS-D/EITHER* SBS-C OR * ANIK - C/3
16	6/16/81	SPACELAB #3, (SSUS-D OF OPPORTUNITY)
17 UP	7/16/81	INTELSAT V, (SSUS-D OF OPPORTUNITY)
17 DOWN	7/19/81	LDEF RETRIEVAL
18	7/29/81	ONE PALLET FOR SPACE PROCESSING, (ONE PALLET OF OPPORTUNITY),(STP-P80-1)
19	9/2/81	5 SPACELAB PALLETS W/IGLOO, PHYSICS AND ASTRONOMY
20	9/30/81	SPACELAB LONG MODULE W/PALLET, LIFE SCIENCE AND ASTRONOMY PALLET
21 UP	10/14/81	(ONE PALLET OF OPPORTUNITY), (MMS OPPORTUNITY), OMS KIT

Figure 32

122

(Figure 33; Figure 34)

There are many JSC facilities that are utilized for crew interface development, i.e. altitude chambers, water immersion facilities, simulators, etc. The following charts present only a few of these that are of passing interest.

The mockup and development lab in building 9A contains the MDF, full-size orbiter mockup, a precision air bearing table and the orbiter trainer when delivered by Rockwell International.

JSC DEVELOPMENT FACILITIES FOR CREW INTERFACES

MOCKUP AND DEVELOPMENT LAB

- MANIPULATOR DEVELOPMENT FACILITY (MDF)
- ORBITER FULL SIZE MOCKUP
- PRECISION AIR BEARING TABLE
- ORBITER TRAINER

Figure 33

124

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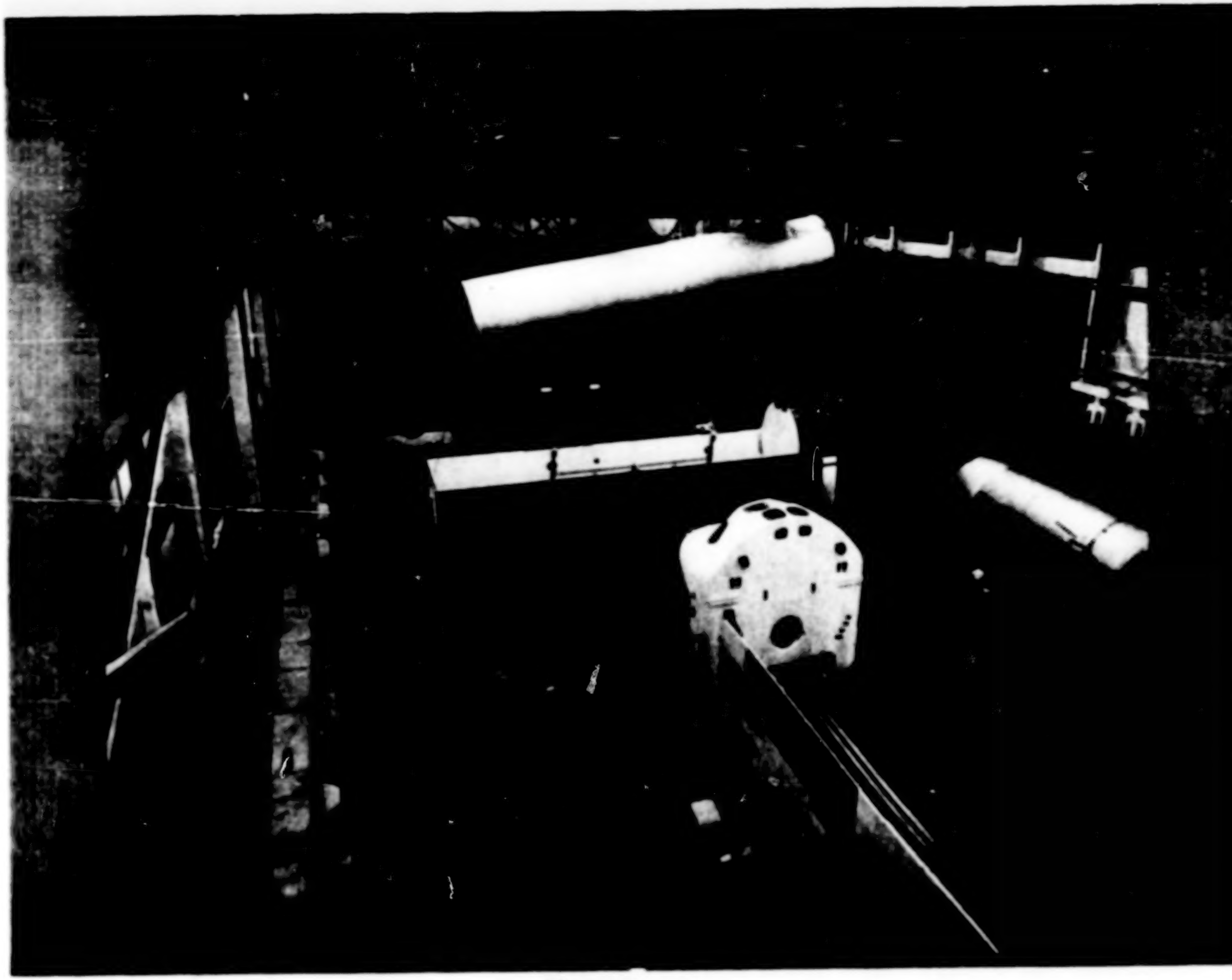

Figure 34

125

126

(Figure 35 ; Figure 36)

The Manipulator Development Facility contains a functional manipulator with the required supporting hardware. It utilizes inflatables that are Helium filled and neutrally ballasted to enable the manipulator arm to move them in and out of the simulated orbiter payload bay.

The arm is also used to move large masses on the adjacent air bearing floor to examine the inertial effects of large payloads when handled with the manipulator. The aft flight deck is duplicated to give realistic external viewing and control station interaction. Representative spacecraft lighting and payload bay fixtures are provided.

MOCKUP AND DEVELOPMENT LAB

- MANIPULATOR DEVELOPMENT FACILITY (MDF)
  - FULL SCALE PAYLOAD BAY
  - AFT FLIGHT DECK WITH FUNCTIONAL CONTROLS AND DISPLAYS
  - CLOSED CIRCUIT TV REPRESENTING FLIGHT CONFIGURATION
  - 50' MECHANICAL ARM - FUNCTIONALLY SIMILAR TO FLIGHT
  - COMPUTER PROGRAMMED TO DUPLICATE FUNCTIONAL CHARACTERISTICS OF FLIGHT SYSTEM - NO VEHICLE DYNAMICS
  - AIR BEARING FLOOR FOR HANDLING LARGE (32,000 LBS) MASSES WITH MANIPULATOR
  - REPRESENTATIVE SPACECRAFT LIGHTING
  - INFLATABLES REPRESENTING VARIOUS PAYLOADS

Figure 35

128

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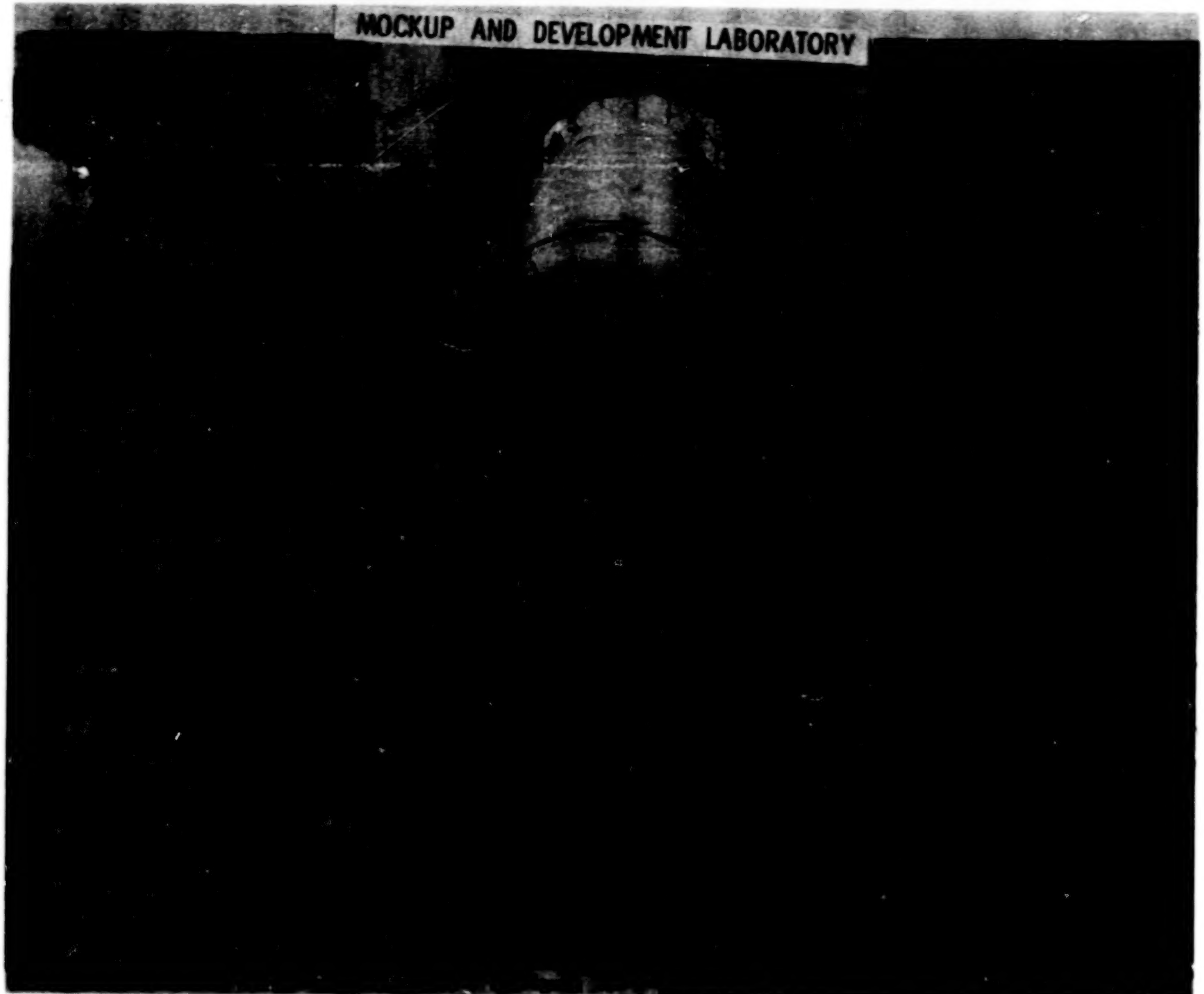

129

(Figure 37;Figure 38)

The full-size crew module is continually updated to the current program configuration. This allows an evaluation of all crew interfaces during the development phase of the cabin. The module has the capability to be rotated to review launch (vertical) conditions as well as on orbit and landing orientations.

The payload bay represents the vehicle structure and will be utilized to review interface issues with various payloads. Some mockup payloads are available and as the program requires, additions to the inventory will be made.

ORBITER FULL SIZE MOCKUP

- CREW MODULE
  - CONFIGURED FOR CURRENT PROGRAM
  - ALL CREW INTERFACES REPRESENTED
  - ABILITY TO CONFIGURE FOR LAUNCH (VERTICAL), ON ORBIT AND LANDING (HORIZONTAL)
- PAYLOAD BAY
  - REPRESENTS DETAIL OF VEHICLE STRUCTURE
  - MAY BE CONFIGURED WITH WIRING OR MISCELLANEOUS EQUIPMENT
  - CAPABLE OF SUPPORTING MOCKUP PAYLOADS IN EXCESS OF 25,000 LBS

Figure 37

132

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ORBITER FULL SIZE MOCKUP

Figure 38

133

CONTENTS

VOLUME I

134

(Figure 3.9)

The orbiter trainer consisting of the crew module and airlock will provide the hi-fidelity configuration required to train all crewmembers. This will be the first program where access to the flight article may not be available to the flight crew before they enter on launch day. Therefore, it must have the same crew interfaces as the flight article to properly train the operators.

ORBITER TRAINER

- HIGH FIDELITY CREW MODULE
- UTILIZED BY FLIGHT CREWS AND GROUND CREWS TO BECOME PROFICIENT IN HANDLING CREW EQUIPMENT AND VERIFYING OPERATING PROCEDURES

Figure 39

135

136

(Figure 40; Figure 41)

The precision air bearing table is shown being utilized for the development of the internal restraint aids or suction cup shoes. Extensive studies over the past few years have shown that the crewmen equipped with suction cups can obtain all the restraint necessary to carry out their tasks within the cabin while under the influence of zero-g. The table has also been utilized for evaluating small maneuvering devices that require very little drag.

PRECISION AIR BEARING TABLE

- PRECISION FLAT SURFACE FOR EVALUATING SMALL MANEUVERING EQUIPMENT
  - GEMINI ASTRONAUT MANEUVERING UNIT
  - SKYLAB MANEUVERING UNITS
  - MANEUVERABLE TV SYSTEMS

Figure 40

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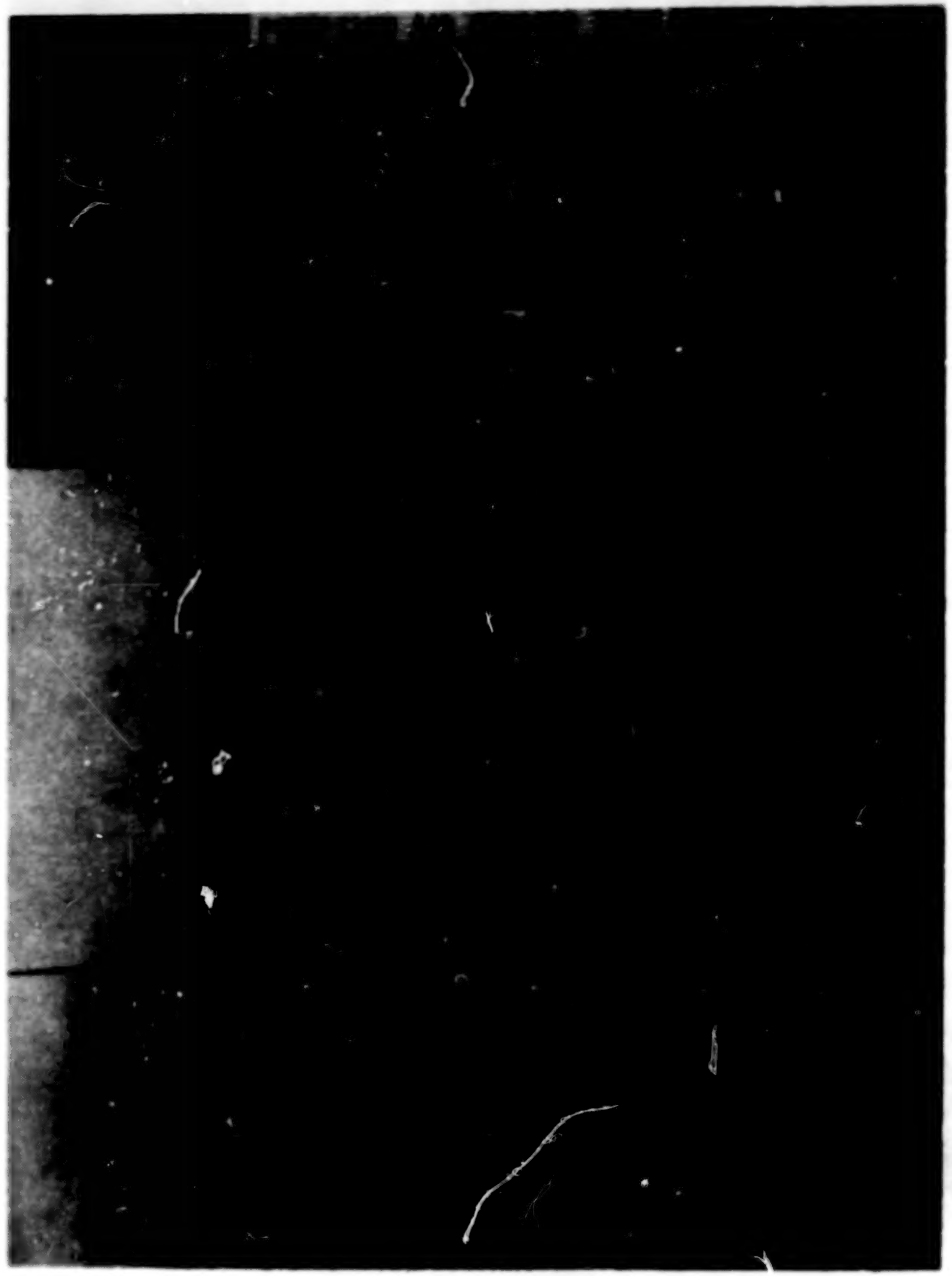

Figure 41

COMMENTS OF GENERAL INTEREST FROM QUESTIONS AND ANSWERS

Shuttle Crew Astronaut Interfaces

1. Approach to Contingency Planning

Extra vehicular activity stands as the present approach toward handling contingenies in orbit. EVA will operate in conjunction with the manipulator and with other provisions such as pyro releases; however the base for planning will involve a man and tools.

2. Present Plans for EVA

No EVA has been formally included in the present Flight plans. However, an EVA activity is in the process of planning relative to OFT Flight #6. For the present, planning for EVA represents planning for a contingency.

3. Interface Control Documents

Over a period which could require a year, the present Volume XIV will be replaced by the two new Interface Control Documents.

DESIGNING STRUCTURES
FOR LARGE SPACE SYSTEMS

R. H. CHRISTENSEN
McDONNELL DOUGLAS ASTRONAUTICS COMPANY

Government/Industry Seminar on Large Space System Technology
January 17-19, 1978

NASA-Langley Research Center

141

142

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CONTENTS

144

1.0 INTRODUCTION

For any new structural s[illegible] codes and standards by which the vehicle will be designed, analyzed, fabricated, tested, and operated will requ[illegible]e a comprehensive set of criteria. It is generally agreed that the formulation of design principles, design criteria, design requirements, or whatever we choose to call them is a tedious and often uninteresting task: (Certainly less interesting than the challenge of conceiving of designs and missions for structural systems of the future.) However, the task must eventually be undertaken, and it would be better done sooner than later.

For many years commercial airplane designers have been able to refer to a detailed set of design requirements covering nearly every aspect of the airplane's structure. In fact, designers are required by Federal regulations to comply with these requirements. Based on the accumulated knowledge of decades of testing and actual service, hammered out by joint industry-Government teams, and kept up to date by periodic revisions, these requirements have contributed significantly to the matchless record of structural safety achieved by the American aircraft industry. Concurrently the U.S. Air Force experienced similar series of events, and eventually their efforts resulted in a set of military specifications for design that are adhered to today.

By contrast, in the relatively youthful space vehicle industry, there is no such uniform governmental regulation. There is only an incomplete set of general recommended guidelines for the design of spacecraft structure. The absence of published requirements (specifications) does not imply, however, that there is no useful information to guide spacecraft designers. On the contrary, NASA, DOD, and industry have invested a great deal of money during the past decade in learning how to build flightworthy vehicles, and there have been several significant breakthroughs and a few expensive catastrophies to mark the educational progress made by the nation's designers.

It is believed that many in industry, either individually or collectively, are willing to assist NASA in their LSST plans. A beginning would be to prepare and respond to a checklist-type questionnaire (but one far more comprehensive than the sketchy one presented here) of all areas that must be accounted for in the structural design of Large Space Systems.

It is not proposed all of this should lead to a set of design specifications, or is it believed that this is even in NASA's plans. The interchange of design information, however, would be welcomed.

DESIGN REQUIREMENTS AND GUIDELINES*

COMMERCIAL AIRCRAFT	FEDERAL AIRWORTHINESS REQUIREMENTS
AIR FORCE VEHICLES	MILITARY SPECIFICATIONS
NASA SPACECRAFT	RECOMMENDED DESIGN GUIDELINES* (SP-8000 SERIES)
LARGE SPACE STRUCTURAL SYSTEMS	SPECIFIC DESIGN GUIDELINES WILL BE* BENEFICIAL

146

2.0 DESIGN CHECKLIST (QUESTIONNAIRE) FOR LARGE SPACE SYSTEMS

The purpose of this paper is to stimulate the interest of the participants at this seminar in responding to a structures design checklist in the LSST areas that have to be accounted for. When this is done, three accomplishments will be evident: (1) the technology deficiencies that probably exist will be identified, (2) such gaps will then point to the "needs" requiring extensive R&D activity, and (3) the results of (1) and (2) above will provide the necessary design guidelines for the Government and industry to follow in support of the Large Space Systems Program.

The design checklist (questionnaire) has been arranged into sections approximately in the order in which the items are accounted for in any preliminary design process. However, no attempt has been made to provide answers.

Although the participants at this seminar are under no obligation to respond to this or their own questionnaire, some response would be appreciated, could be held in confidence, and could assist NASA in both its short- and long-range technology operating plans for the LSST program.

Within the sections of the questionnaiee are included some of the important disciplines for the design of any new system. The list undoubtedly will suggest other areas overlooked, and it is proposed that both Government and industry expand upon it. Directing the combined thinking of structures and materials specialists within NASA, DOD, and industry with such a format will result in more meaningful and timely discussions. This should reduce the number of unintegrated ad-hoc technology investigations that often are unnecessarily performed.

RESPONSE FROM THE QUESTIONNAIRE YOU RECEIVE WILL:

1. IDENTIFY TECHNOLOGY DEFICIENCIES

2. POINT TO R&D ACTIVITIES REQUIRED

3. PROVIDE DESIGN GUIDELINES TO SUPPORT THE LSST PROGRAM FROM COMPLETION OF 1. AND 2.

148

PROOF OF DESIGN

LOAD ANALYSES

THERMAL ANALYSES

STRUCTURAL ANALYSES

STABILIZATION AND CONTROL OF FLEXIBLE LSS

MATERIAL CHARACTERIZATION TESTS

STRUCTURAL TESTS

THERMAL ANALYSES

THERMAL ANALYSES SHOULD BE MADE OF THE INDUCED THERMAL ENVIRONMENT AND OF THE VEHICLE SYSTEM RESPONSE TO THIS ENVIRONMENT. PLEASE COMMENT ON THE ADEQUACY OF CURRENT THERMAL ANALYSIS TECHNIQUES IN EVALUATING THE THERMAL EFFECTS ON STRUCTURAL MATERIALS, STRUCTURAL COMPONENTS AND THEIR ASSEMBLY, AND INSULATION MATERIALS OF ON-ORBIT LARGE STRUCTURAL SYSTEMS.

ANSWER . . . .

150

STABILIZATION AND CONTROL OF FLEXIBLE LARGE SPACE STRUCTURAL SYSTEMS

STATIC AND DYNAMIC ANALYSES SHOULD ACCOUNT FOR VEHICLE AND CONTROL SYSTEM STIFFNESS DISTRIBUTIONS, STRUCTURE/CONTROL SYSTEM INTERACTIONS, INTERFACE WITH THERMAL CHARACTERISTICS, AND STABILITY MARGINS.

A. WHAT MECHANISMS CAN BE EMPLOYED TO PROVIDE A PRESCRIBED SURFACE OF A LARGE SPACE STRUCTURE WHICH IS SUBJECT TO:
   1. INTERNALLY GENERATED FORCES (THERMAL, POINTING CONTROL TORQUES)
   2. EXTERNALLY GENERATED FORCES (GRAVITY GRADIENT TORQUE, AERODYNAMIC SOURCES)
   3. CONSTRUCTION TOLERANCES

B. WHAT MEASUREMENT SYSTEM (INSTRUMENTATION SENSORS) CAN BE EMPLOYED TO DETERMINE SURFACE TOLERANCE, DISTORTION AND VIBRATION FOR LARGE SPACE ANTENNA SURFACES?

ANSWER(S) . . . .

3.0 SOME SUGGESTED R&D PROJECTS

The trend in advanced spacecraft structures will continue to be toward simplicity, reliability, and cost effectiveness. Much remains to be done; for example, significant potential payoffs in structural design must be correlated with solutions to a variety of problem areas.

Some of the general problem areas were identified by category compiling the initial "Design Checklist" of this paper; they are noted below. This is a start; yet, many others should be added and eventually expanded into detailed work statements.

- Need to Improve Flightworthiness

  Structural integrity must be improved to provide flightworthy vehicles for the kinds of advanced, highly demanding missions that are now being planned.

- Need to Satisfy Cost Constraints

  The structural systems must be improved (e.g., through standardization, simplified fabrication, and minimized testing and paperwork) to provide a competitive product within cost constraints (e.g., space shuttle payloads designed and built within NASA's established cost targets).

- Need to Assess New Loads and Environments

  The effects of new loads and environments have not yet been determined for future planned missions; and, these data are urgently needed for design of efficient, low-cost structural systems.

- Need to Improve Mission Performance (Including Life)

  The structural systems must be improved to meet the stringent mission performance requirements of advanced spacecraft vehicles. These requirements include items such as: (1) extended life times in space with subsequent reuse; and (2) very frequent take offs and landings in landing fields having area restrictions.

- Need to Apply Technology to Preserving the Ecology

  The preservation of the natural environment and improvements in the quality of life are high priority goals for current and future aerospace programs.

- Need for Design Criteria

  Criteria and guidelines have been formulated for a specific technology, although it is reasonably well established and developed.

Similarly, assessment of NASA's role reveals a number of critically urgent research programs in which structural system design studies and criteria would provide immediate benefits. Although it is clearly NASA's imperative to designate such programs, the industry is available and ready to assist NASA in many ways.

151

152

4.0 CONCLUDING REMARKS

The participants at this seminar have been assembled for discussions on questions of NASA interest. Certainly the collective knowledge of the participants, willing to assist NASA, would support the needed technology to develop successful large space systems. It is suggested that one initial approach would be to ask and answer questions similar to the ones presented in the checklist questionnaire of this paper.

Finally, a plea is made to the structures designer and materials specialist to work closely together in the early design stages if rapid advances are to be made in the future. Unless this occurs it is likely that the structural problems, due to the challenge of increasingly difficult environments, will take much longer to be solved.

There are requirements to find ways to reduce cost and to reliably process and fabricate structure from new material systems. Means of improving weld reliability, inspection and maintainability/refurbishment, etc. should also be sought. In all of these things, the engineers should seek ways to develop improved test coupon designs and test techniques to give more reliable data for the use of the designer.

ORBITING DEEP SPACE RELAY STATION

(ODSRS)

DSN FEASIBILITY STUDY REPORT

JANUARY 18, 1978

TOM THORNTON/JOHN HUNTER

JET PROPULSION LABORATORY

153

6553-2978

153

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1-11-78

154

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ORBITING DEEP SPACE RELAY STATION
RATIONALE

FUTURE TRACKING REQUIREMENTS

- TELEMETRY DATA RATES
- NAVIGATION ACCURACY REQUIREMENTS
- SIMULTANEOUS OPERATIONS OF SEVERAL MISSIONS

ADVANTAGES OVER EARTH BASED STATIONS

- FREQUENCIES EXCEEDING EARTH ATMOSPHERIC WINDOW
- COLLECTOR AREAS EXCEEDING REASONABLE EARTH BASED STRUCTURE
- SHUTTLE MAKES ORBITING STATION CONCEPT VIABLE

FUTURE THREATS TO EXISTING STATIONS

- POLITICAL AND ECONOMIC FACTORS LIKELY TO IMPEDE OVERSEAS STATION OPERATIONS
- MILITARY ACTIVITIES COMPETE WITH TRACKING ACTIVITIES AT GOLDSTONE

Figure 1

156

ORBITING DEEP SPACE RELAY STATION
STUDY OBJECTIVES

DEVELOP LONG RANGE (1985-1995) DEEP SPACE TRACKING PLAN

- ASSESS TRACKING REQUIREMENTS OF PROBABLE MISSIONS
- SYNTHESIZE END TO END TRACKING SYSTEM OPTIONS TO MEET THESE REQUIREMENTS - ASSUME TERRITORIAL U.S. GROUND STATIONS AND ODSRS
- DETERMINE OPTIMUM COST/BENEFIT CONFIGURATION

DEFINE TECHNOLOGY DEVELOPMENT REQUIREMENTS

PROVIDE PRELIMINARY IMPLEMENTATION PLAN FOR SELECTED CONFIGURATION

- INCLUDE PHASEOVER FROM PRESENT CONFIGURATION

Figure 2

ORBITING DEEP SPACE RELAY STATION

STUDY CONSTRAINTS

- SYSTEM OPERATIONAL BY 1985 - 1987
- SYSTEM LIFETIME TEN YEARS WITHOUT MAJOR REFURBISHING
- AS A MINIMUM, SYSTEM MUST MEET CURRENT DEEP SPACE TRACKING CAPABILITIES
- CREDIBLE EXTENSIONS OF KNOWN TECHNOLOGY
  - DEMONSTRATE ON EARTH PRIOR TO LAUNCH
  - ASSEMBLE AND CHECKOUT IN LOW EARTH ORBIT
- TECHNOLOGY COMPATIBLE WITH SYSTEM PERFORMANCE GROWTH IN FUTURE
- LIFE-CYCLE COST

Figure 3

ORBITING DEEP SPACE RELAY STATION
DSN FEASIBILITY STUDY - 1977

OBJECTIVES

- DETERMINE TECHNICAL FEASIBILITY OF AN ODSRS
- IDENTIFY TECHNOLOGY DEVELOPMENTS
- PROVIDE BASIS FOR OBTAINING MANAGEMENT SUPPORT FOR DESIGN STUDY AND TECHNOLOGY DEVELOPMENT

CONSTRAINTS

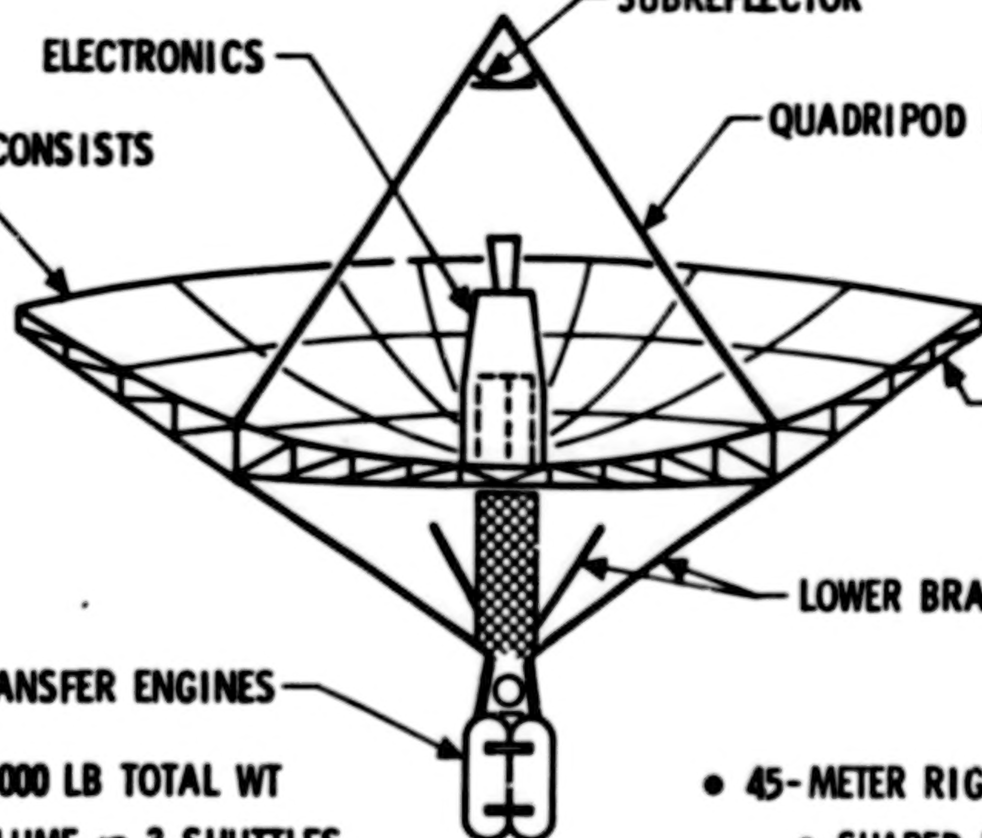
- POINT DESIGN TO SHOW FEASIBILITY - SYSTEM NOT OPTIMIZED
- NO ODSRS UNIQUE CHANGES TO SPACECRAFT OR GROUND
- PERFORMANCE EQUAL TO EXISTING 64-METER STATIONS
- LISTEN ONLY

Figure 4

ORBITING DEEP SPACE RELAY STATION

DSN FEASIBILITY STUDY - 1977

CONFIGURATION



- 13,000 LB TOTAL WT
- VOLUME ≃ 3 SHUTTLES

- 45-METER RIGID DISH
  - SHAPED PARABOLOID MAIN REFLECTOR
  - CASSEGRAIN FEED

159

Figure 5

ORBITING DEEP SPACE RELAY STATION

DSN FEASIBILITY STUDY - 1977

TELECOMMUNICATIONS SYSTEM

SPACECRAFT TO ODSRS LINK

- S, X, OR K-BAND
  - ANY 2 SIMULTANEOUSLY
- MINIMUM ODSRS ANTENNA SIZE: 45 METERS
- MINIMUM ODSRS SYSTEM NOISE TEMPERATURE: 11.5°K

ODSRS TO GROUND LINK

- K-BAND
  - 2 CHANNELS
- 0.5 WATT TRANSMITTER
- 1-METER ODSRS ANTENNA
- 3-METER GROUND ANTENNA
- 250 KB/S TELEMETRY CAPABILITY AT 10^{-5} BER
- MAINTAIN PHASE COHERENCY FOR DOPPLER DATA
- "BENT PIPE" MODE - NO DATA PROCESSING BY ODSRS

Figure 6

ORBITING DEEP SPACE RELAY STATION
DSN FEASIBILITY STUDY - 1977
CONCLUSIONS

TECHNICAL

- FEASIBLE WITH REASONABLE EXTENSIONS OF EXISTING TECHNOLOGY
- 4 TECHNOLOGY DEVELOPMENTS REQUIRED
    - ASSEMBLY IN SPACE
    - 10-YEAR CRYOGENIC PREAMPLIFIERS
    - 10-YEAR HIGH PRECISION ATTITUDE CONTROL
    - 10-YEAR POWER SYSTEM

SCHEDULE

- 1986 LAUNCH FEASIBLE FOR ASSUMED DESIGN
- DESIGN STUDY 1978
- TECHNOLOGY DEVELOPMENT START 1979

Figure 7

161

162

ANGULAR RESOLUTION: OPTICAL VS RADIO ASTRONOMY MAPS (Figure 8)

Very Long Baseline Interferometry (VLBI) techniques have been used by radio astronomers over the last decade to obtain maps of celestial radio sources at previously unrealizable levels of angular resolution. Angular resolutions have reached below 10^{-3} arc-seconds, or about three orders of magnitude smaller than the resolution achieved with optical photographs or conventional radio interferometers. Satellite-borne VLBI terminals could be used to provide maps of compact celestial radio sources with finer resolution, less ambiguity, and more efficiency than earth-bound VLBI techniques. These maps and their time variability would help unravel the physical processes that govern some of the most enigmatic classes of celestial objects.

ANGULAR RESOLUTION: OPTICAL vs RADIO ASTRONOMY MAPS

$$(\text{ANGULAR RESOLUTION} \approx \frac{\text{WAVELENGTH}}{\text{TELESCOPE DIAMETER}})$$

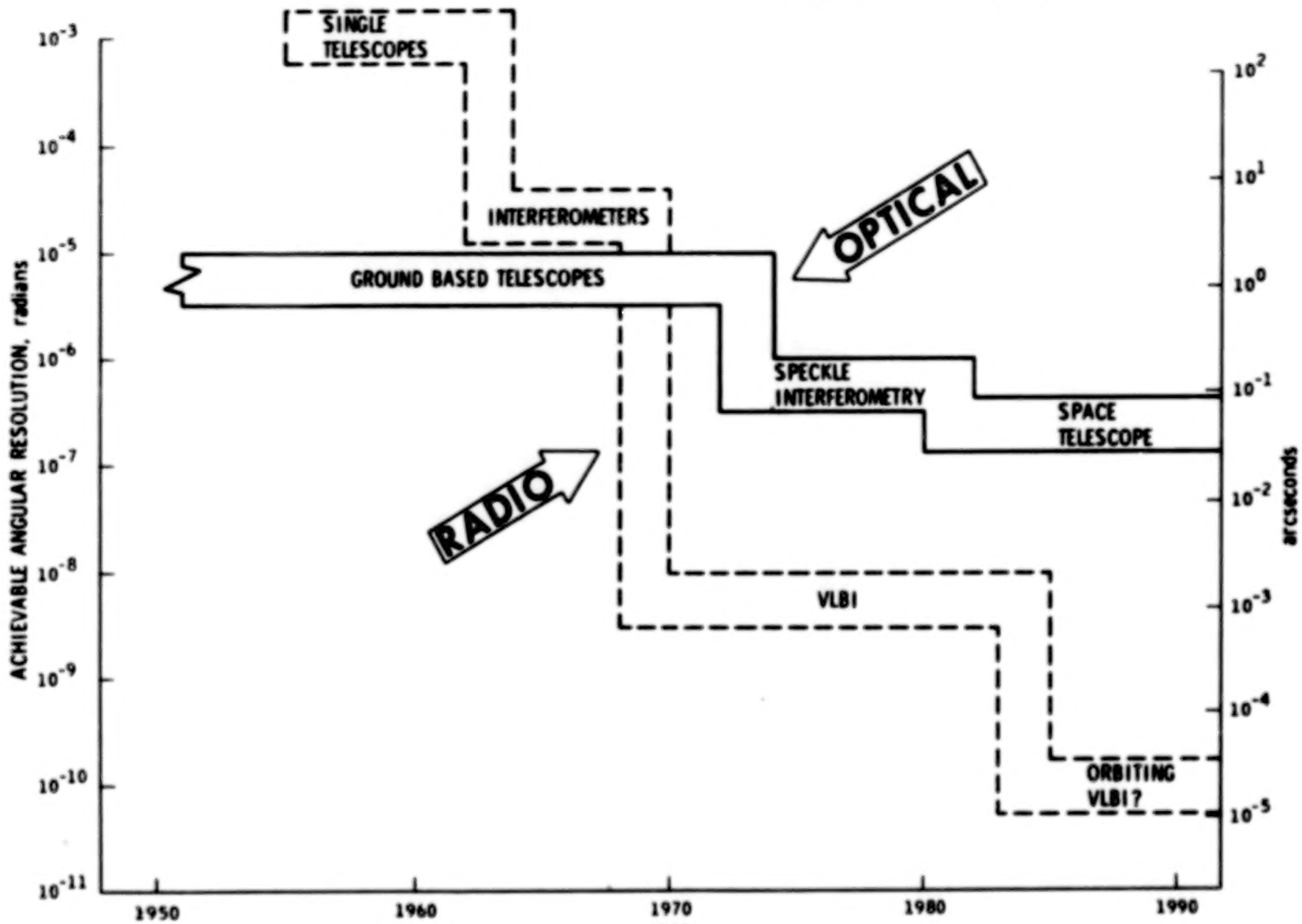


Figure 8

163

164

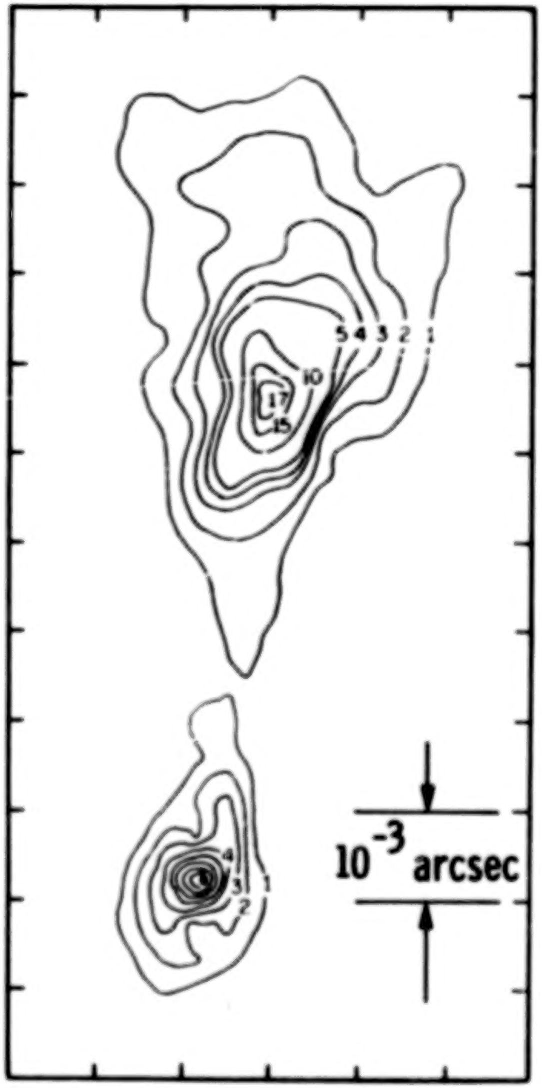
ORBITING VLBI: HIGH RESOLUTION MAPS OF CELESTIAL RADIO SOURCES (Figure 9)

An orbiting VLBI system would be sensitive only to celestial objects that emitted enormous amounts of power from small angular elements of the sky, and so would probe regions which were unusual and excited. These regions are very far out of thermodynamic equilibrium, many of them are variable on short time scales, and their fundamental physical processes are unknown. Proper monitoring of these objects will require frequent observations at very high angular resolution. Such observations could certainly best be performed by means of an orbiting VLBI terminal.

The principal scientific objectives of an orbiting VLBI system would be to investigate:

- apparent faster-than-light phenomena in quasars and galactic cores
- the general physical processes that govern and the relationships among quasars, BL Lacertae objects, and galactic cores
- the maser phenomenon exhibited by clouds of interstellar molecules and the importance of these clouds in stellar formation
- the physics of energetic stars (pulsars, x-ray stars, flare stars)

ORBITING VLBI: HIGH RESOLUTION MAPS OF CELESTIAL RADIO SOURCES



- CELESTIAL OBJECTS OF INTEREST:

  QUASARS
  BL LACERTAE OBJECTS
  GALACTIC CORES
  INTERSTELLAR MASERS
  EXOTIC STARS

- ACHIEVABLE ANGULAR RESOLUTIONS ARE > 10^3 SMALLER THAN BY OTHER OPTICAL OR RADIO TECHNIQUES

Figure 9

166

THE VLBI TECHNIQUE (Figure 10)

VLBI observations require a pair of antennas to simultaneously observe the same celestial source. These antennas are generally widely separated, often up to transcontinental or intercontinental distances for earth-bound VLBI, as the minimum size angular structure which can be resolved is inversely proportional to the distance (baseline) between the two antennas. With orbiting VLBI, one antenna may be in orbit with the other earth-bound, or both may be in orbit. At each antenna the signals being received are heterodyned to lower frequencies with the phase stability of the receivers and timing of the digital sampling controlled by atomic clocks. Signals from the two sites are brought to a common location by direct transmission or transported tape recordings and then are cross-correlated.

THE VLBI TECHNIQUE

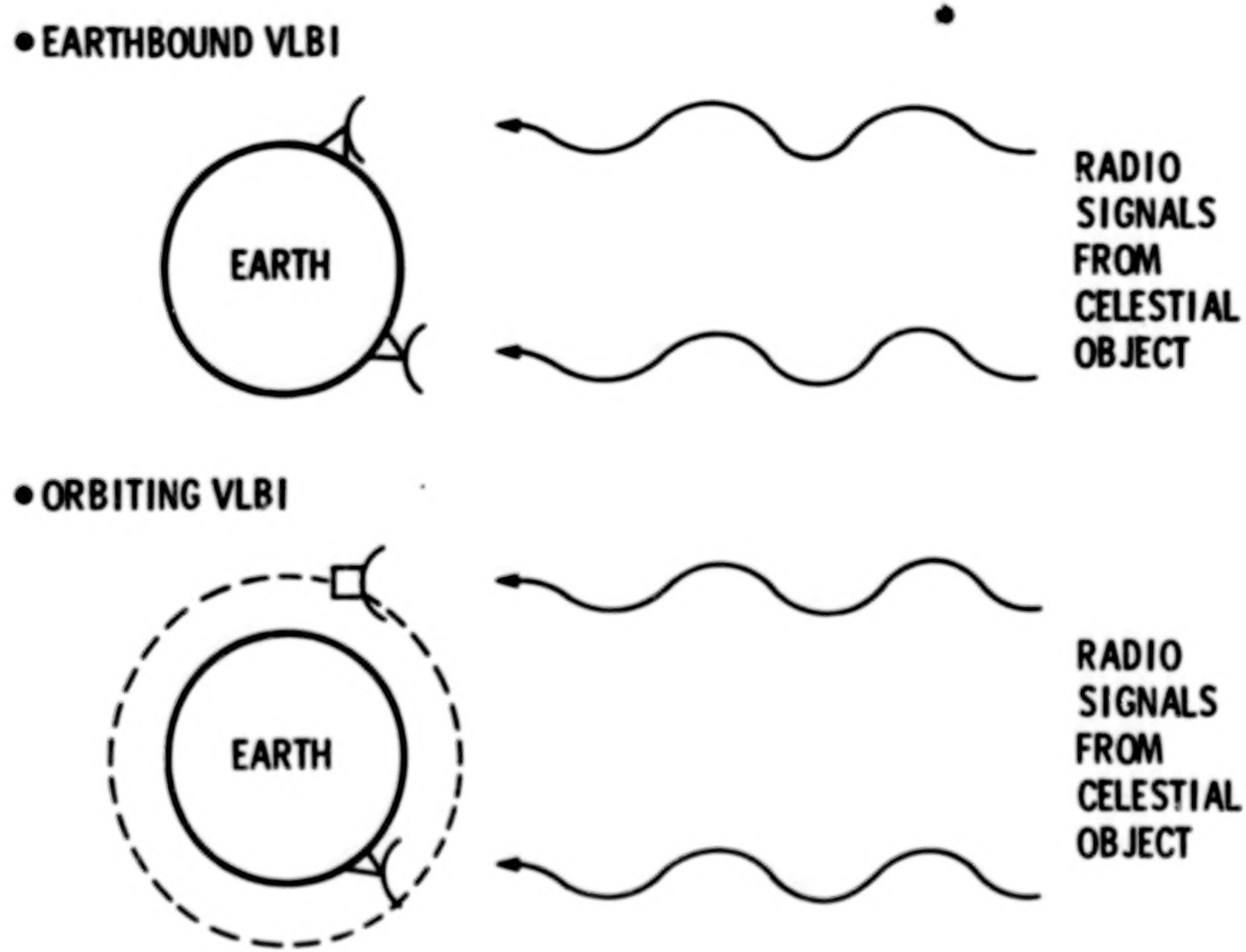


Figure 10

168

ORBITING VLBI: SYNTHESIZING ANTENNAS LARGER THAN THE EARTH (Figure 11)

For any particular measurement time, two antennas performing VLBI measurements act in effect like two small elements of a much larger antenna. If the geometry of the baseline with respect to the radio source direction is varied, different elements of the imaginary large antenna can be sampled. The more completely the large antenna is synthesized, the more reliably an accurate map of the source can be reproduced. The resulting map would possess angular resolution corresponding to that expected from the synthesized antenna (angular resolution $\approx$ wavelength/antenna diameter). With orbiting VLBI, the synthesized antenna may have a diameter several times larger than the earth.

ORBITING VLBI: SYNTHESIZING ANTENNAS LARGER THAN THE EARTH

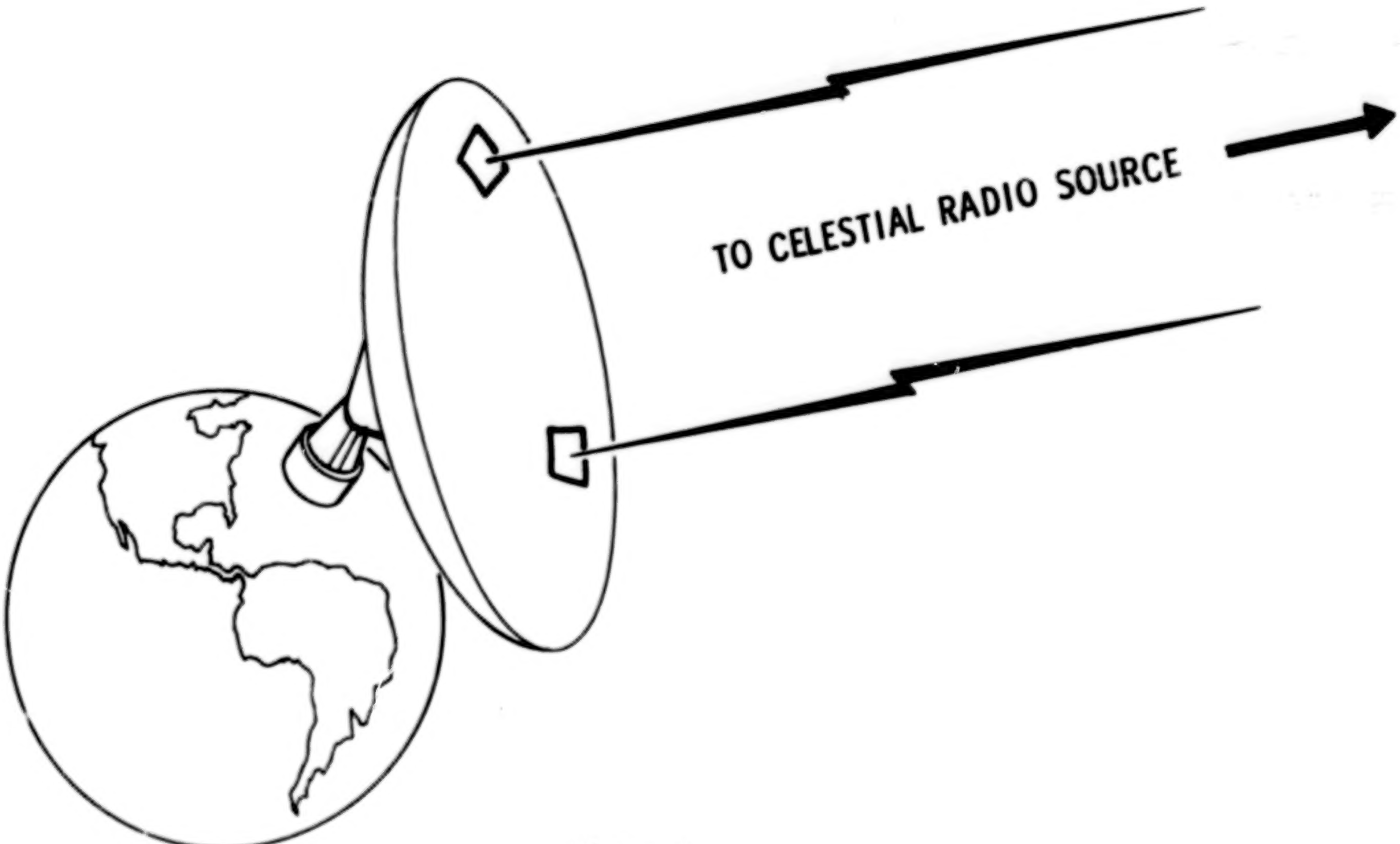


Figure 11

169

170

ADVANTAGES OF AN ORBITING VLBI OBSERVATORY (Figure 12)

The advantages of orbiting VLBI over earth-bound VLBI are threefold:

- Less ambiguous maps - The geometric variations available with satellite-borne antennas far surpass the variations of earth-bound antennas alone.
- Greater angular resolution - Baseline lengths with orbiting VLBI are essentially unbounded, whereas earth-bound VLBI is limited by the earth's diameter.
- More rapid mapping - For near-earth satellites, geometric coverage is much more rapid than for a pair of earth-bound antennas.

ADVANTAGES OF AN ORBITING VLBI OBSERVATORY

- LESS AMBIGUOUS MAPS

- GREATER ANGULAR RESOLUTION

- MORE RAPID MAPPING

Figure 12

ORBITING VLBI: HIGH RESOLUTION MAPS OF CELESTIAL RADIO SOURCES (Figure 13)

Although orbiting VLBI may be performed from the Space Shuttle with an antenna that both deploys and collapses upon command, the wealth of information to be gathered in years of observing provides a compelling reason to place an orbiting VLBI system on a free-flyer spacecraft. In addition, eventual demands for greater angular resolution would call for orbital altitudes not within Shuttle's capability.

The free-flyer spacecraft would initially be carried into orbit by Shuttle with antenna deployment occurring after release. Observations would be performed simultaneously with one or more ground antennas or another free-flyer VLBI observatory, with data from the orbital VLBI system(s) being transferred to a ground site in real time via the Tracking and Data Relay Satellite System (TDRSS). If the data from two VLBI antennas can be brought together in real time, correlation can also occur in real time. Otherwise, data will be recorded on tape and correlated at a later time.

The orbiting VLBI system requires a 30- to 60-meter-diameter deployable parabolic antenna, which will probably employ a Cassegranian subreflector so that the receiver instrumentation can be conveniently co-located at the base of the antenna with the remainder of the spacecraft. The subreflector would be steerable to point at various feed horns mounted at the antenna base so that different frequency bands between 1 and 22 GHz could be covered. Sensitive low noise receivers are required which may necessitate cryogenic cooling. Following amplification, the signals are heterodyned to lower frequencies where they are digitally sampled at a rate ranging between 4 and 120 Mbits/sec. Frequency conversion and digital sampling are controlled by a hydrogen maser atomic frequency standard, which may be on board the spacecraft, or whose signal may be relayed from the ground.

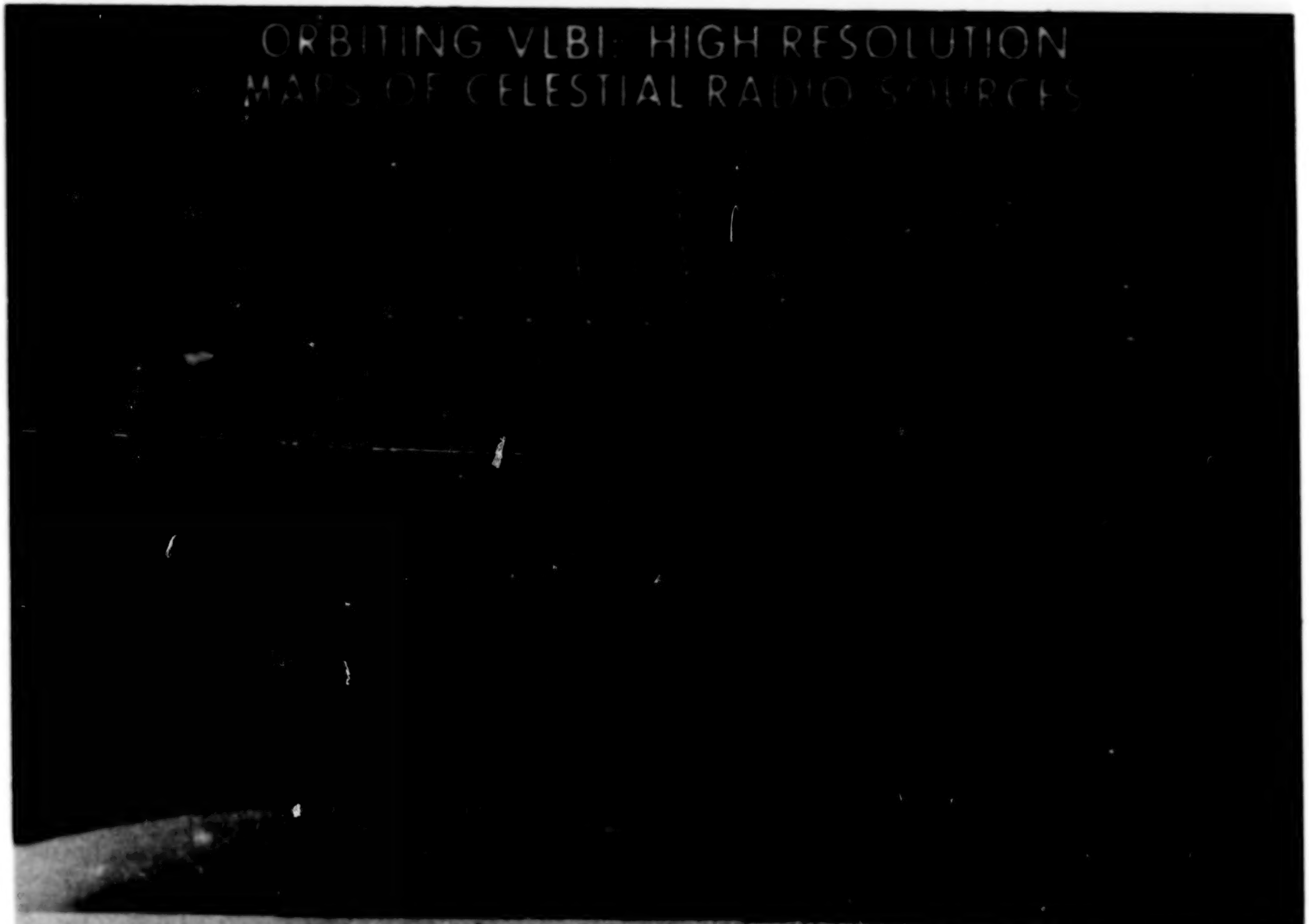


Figure 13

173

174

TECHNICAL CHALLENGES (Figure 14)

A useful orbiting VLBI observatory could be built with existing state-of-the-art technology. A low sensitivity VLBI system utilizing present technology has been proposed as a Spacelab experiment on Shuttle. However, a high sensitivity free-flyer VLBI observatory will require moderate technology development in several areas.

TECHNICAL CHALLENGES

- DEPLOYABLE PARABOLIC ANTENNAS
  - 30-60 METER DIAMETER
  - LESS THAN 2 MILLIMETERS SURFACE TOLERANCE
- MOMENTUM WHEEL ATTITUDE CONTROL SYSTEM
  - FEW ARC-SECOND ACCURACY
- SOLAR POWER DESIGN
- HYDROGEN MASER ATOMIC FREQUENCY STANDARD
- CRYOGENIC RECEIVERS

Figure 14

176

COMMENTS OF GENERAL INTEREST FROM QUESTIONS AND ANSWERS

Orbiting Deep Space Relay Station and VLBI for Radio Astronomy

Cryogenic Technology

These space system concepts identify a requirement to improve technology for cryogenic cooling; however, the advances foreseen do not appear distant from the present level of capability.

HIGH RESOLUTION SOIL MOISTURE RADIOMETER

T. T. WILHEIT, GODDARD SPACE FLIGHT CENTER

177

The experiment concept being examined here is a high resolution soil moisture radiometer. A very low resolution (100 km) soil moisture radiometer was flown on Skylab (S-194) and systems under consideration for the 1979-1983 time frame could be considered medium resolution (10-20 km). By high resolution we mean 1 km.

(Figure 1)

In planning for the time scale in which large structures might reasonably be built, one cannot be limited to well articulated requirements; it is often necessary to anticipate future requirements. The requirement of measuring soil moisture at 1 km resolution is such a case. There are a variety of good reasons to anticipate such a requirement and at least one source, the World Meteorological Organization (WMO) has expressed such a need.

PROJECTED REQUIREMENT FOR LATE ' 80's
SOIL MOISTURE AT 1 KM RESOLUTION

- MEETS MINIMUM WMO REQUIREMENTS (100M -1 KM)
- THE SMALLER MORE STRUCTURED WATERSHEDS ARE SOURCE REGIONS FOR MAJOR WATERSHEDS
- PERMITS SOME WORK IN U.S. AND EUROPE
- SCALE SIZE OF CONVECTIVE RAINFALL
- SCALE SIZE OF MAJOR AGRICULTURAL FIELDS (QUARTER SECTION)

USEFUL FOR: RUNOFF PREDICTIONS
MOISTURE BUDGET MODELS
CLIMATE
CROP YIELD

Figure 1

180

(Figure 2)

Two committees under the aegis of the National Academy of Science have stated that the use of long wavelength ($\lambda \geq 20$ cm) passive microwave measurements appears at present to be the most reasonable approach to the remote measurement of soil moisture. The penetration depth for the soil moisture measurement is about 1/20 of the free space wavelength. Thus, long wavelengths, 20 cm or longer, are required in order to make agriculturally meaningful measurements. Two frequencies will be considered; 1.4 MHz which is presently a radio astronomy band and 611 MHz which has been proposed as a future radio astronomy band.

APPROACH

- LOW FREQUENCY MICROWAVE RADIOMETER
  - PENETRATION INTO SOIL
  - DEMONSTRATED RELATIONSHIPS
- POSSIBLE FREQUENCIES
  - 1.4 GHZ RADIO ASTRONOMY ROAD 27 MHZ WIDE
  - 611 MHZ PROPOSED RADIO ASTRONOMY BAND 6 MHZ WIDE
- ELECTRICALLY SCANNED PHASED ARRAY ANTENNA
  - ELIMINATES MECHANICAL SCAN
- MULTIBEAM - PUSHBROOM SCAN
  - INTEGRATION TIME (S/N) REQUIRES IT

Figure 2

CONTENTS

VOLUME I

182

(Figure 3)

This slide is a schematic representation of the antenna concept being considered here. Multiple beams and lack of mechanical scanning are essential to meeting the objectives of this system. The antenna is made of a series of waveguide elements with radiating slots. The waveguide dimensions and the slot spacing determines the angle of the beam with respect to the waveguide; the relative phasing among the elements determines the beam location in the plane perpendicular to the waveguide. The signal from each element is amplified by a low noise amplifier and the signals are combined in a beam-forming matrix which consists of power dividers and delay lines and has one output port for each beam formed.

PUSHBROOM PHASED ARRAY

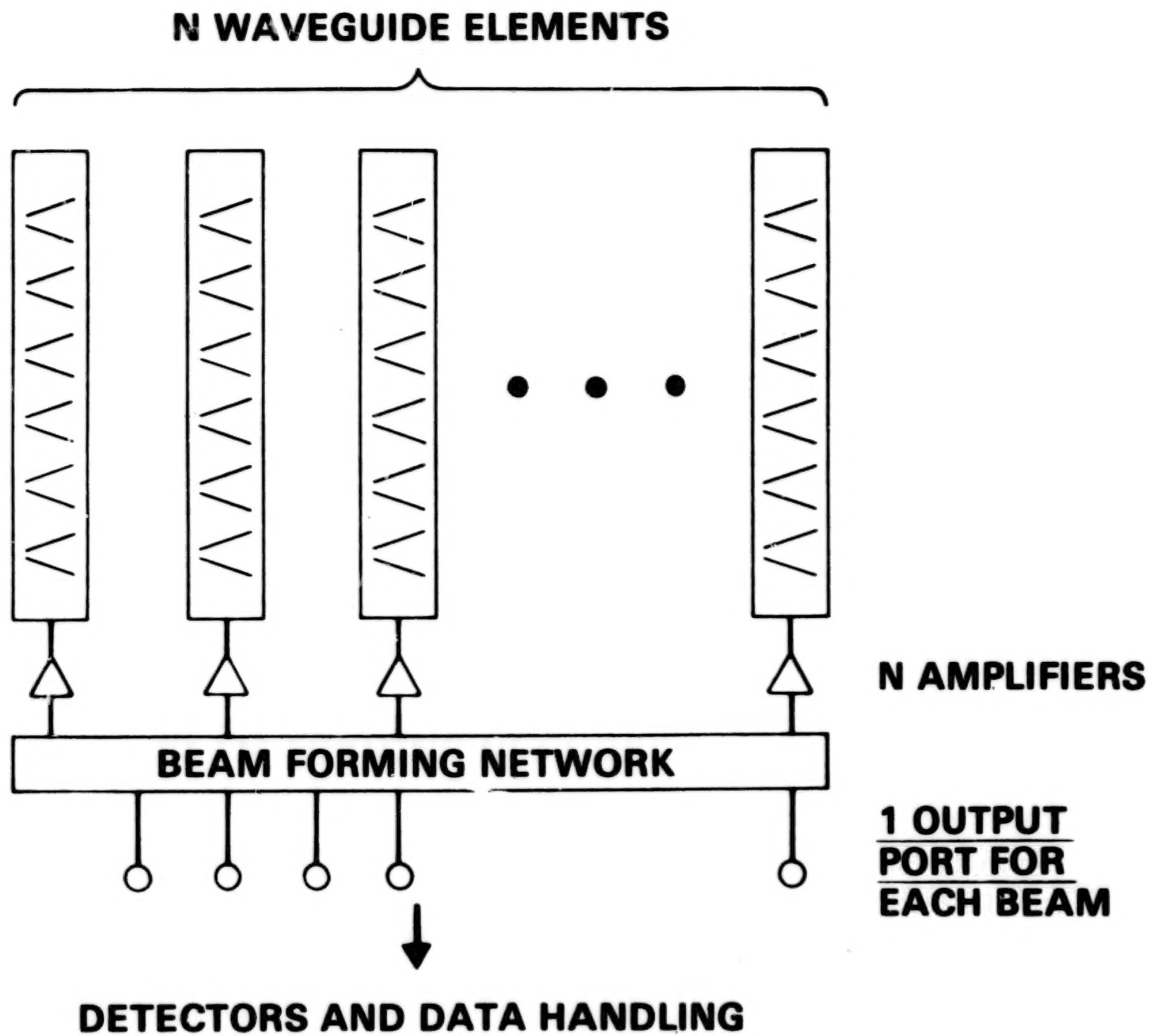


Figure 3

183

184

(Figure 4)

We assume an orbit of 388 km altitude and 50^{o} inclination. This would give a three-day repeat cycle which would be appropriate for soil moisture measurements. We will also carry in parallel two assumed frequencies, 1400 and 611 MHz. With these assumptions, an antenna size of 100 meters at 1400 MHz or 230 meters at 611 MHz is required for a resolution of 1 km at the Earth's surface. Such a size is much larger than can be fit into the Shuttle, even assuming deployment. It will clearly have to be carried up to orbit in several Shuttle launches and assembled on orbit. One approach would be to build the waveguides themselves in space.

ANTENNA SIZES

F=1.4 GHZ (27 MHZ BW)
611 MHZ (6 MHZ BW)

ASSUME ORBIT 388 KM ALTITUDE
50⁰ INCLINATION
(3 DAY REPEAT)

DESIRED RESOLUTION: 1 KM

ANTENNA SIZES: 100 M, 230M

BUILD SLOTTED WAVEGUIDES IN ORBIT

(Figure 4)

(Figure 5)

Complete coverage through the agriculturally important portion of the globe in a reasonably short time scale is a requirement on any soil moisture mission. The orbit has been chosen to give a three-day repeat cycle which will fulfill the timeliness portion of the requirement. This, in turn, specifies the swath width needed. Specifically, with the orbit being considered here, a swath width of 455 km will yield total earth coverage every three days between latitudes 20° and 54° both north and south. This can be accomplished with a $\pm 30^{\circ}$ scan. This is sufficiently near nadir that all the data will have very nearly the same resolution and reflect very nearly the same physics.

SWATH WIDTH REQUIREMENT

3 DAY REPEATING ORBIT
50° INCLINATION
455-KM SWATH ($\pm$ 30° SCAN)
COMPLETE THREE-DAY COVERAGE
30-54° LATITUDE
COVERS:

UNITED STATES
THE UKRAINE
SOUTHERN CANADA

Figure 5

187

188

(Figure 6)

Since typical low orbiting satellite ground speeds are about 7 km/sec a 1 km resolution pushbroom beam implies an integration time of 1/7 sec. The system bandwidth is limited by beam smear. That is, in a dispersive structure like a phased array antenna the beam moves as a function of frequency so that the net beam is the convolution of the individual single frequency beams over the entire bandwidth. If the motion of the beam across the frequency range is comparable to the single frequency beamwidth, there will be a degradation of the resolution performance. Thus, beam smear provides a limitation on bandwidth. In the absence of a specified design, one can approximate this effect on a simple coherence length basis. That is, the bandwidth must be small compared to the antenna size divided by the speed of light. On this basis we will use 500 and 200 kHz for the two bandwidths.

SIGNAL/NOISE

PUSHBROOM SCAN, 1 KM RESOLUTION ALLOWS
1/7 SEC INTEGRATION TIME

BEAM SMEAR LIMITS BANDWIDTH

$$\text{BANDWIDTH} << \frac{\text{SPEED OF LIGHT}}{\text{SIZE OF ANTENNA}}$$

I.E., BW << 3 MHZ, 1.3 MHZ .

FOR CALCULATION USE 500 KHZ, 200 KHZ

Figure 6

190

(Figure 7)

Carrying these numbers through with reasonable assumptions for the noise temperature of transistor amplifiers in this frequency range we arrive at 4° and $7^\circ K$ noise estimates. Since about 1° is needed, this is excessive. The solution is to use many narrow frequency band filters and multiple detectors. This requires a great deal of repetition of components which are inexpensive and simple to mass produce.

Such a system would produce many, noisy images which are somewhat displaced one from another and could be rectified and averaged in software either in space or on the ground. If the beam forming matrix can be made non-dispensive so that there is no cross-track displacement, then the images could be rectified by appropriate delays in the integrate and dump filters. The timing delays needed would be of the order of tens to hundreds of milliseconds and thus trivially attainable. Depending on exactly how the system was implemented, the resulting data rate would be of the order of 50 to 500 KB/S which presents no engineering problems.

SIGNAL/NOISE CONTINUED

WITH REASONABLE RECEIVER AND SIGNAL TEMPERATURE ASSUMPTIONS

NEΔT $\approx$ 4^{0}K, 7^{0}K

TOO LARGE

SOLUTION :

MULTIPLE NARROW BAND CHANNELS

RECTIFY IMAGES IN DATA PROCESSING

LIMITING NEΔT $\approx$ 1^{0}K

DATA RATE

50-500 KB/S

Figure 7

192

(Figure 8)

There are many problems to be studied in such a system. We have received some funding to begin these studies and are writing a statement of work to study some specific instrument problems as indicated.
Hopefully, such a study will reveal any hidden pitfalls and provide approaches to cope with them. Certainly, it will provide a sound basis for comparing this approach with others in any trade-off studies.

TECHNICAL STUDIES REQUIRED

PROBLEMS INHERENT TO SIZE OF STRUCTURE

- CONFIGURATION
- MAINTENANCE OF TOLERANCES
- ATTITUDE CONTROL
- ORBITAL MECHANICS OF LARGE STRUCTURES
- ASSEMBLY TECHNIQUES

SPECIFIC INSTRUMENT PROBLEMS

- CALIBRATION
- IMPLEMENTATION OF BEAM FORMING (BUTLER) MATRIX
- MASS PRODUCTION OF REPEATED ELEMENTS
  - E.G., SLOTTED WAVEGUIDES
    - MICROWAVE INTEGRATED CIRCUIT RECEIVERS

PRELIMINARY DESIGN STUDY

- WEIGHT
- POWER
- COST
- ETC.

Figure 8

194

COMMENTS OF GENERAL INTEREST FROM QUESTIONS AND ANSWERS

High Resolution Soil Moisture Radiometer

1. High Resolution Requirements and Disturbing Effects

To an orbiting radiometer, the eastern portion of the United States presents a hetrogenous surface comprised of land parcels which approximate 1 km^2. Such parcels represent individual farms, local watersheds, etc. useful data projections require working with elements of this size. Soil moisture content influences the dielectric constant of the soil; detection becomes an emissivity rather than a thermal effect. Scanning from near verticle minimizes the impact of surface roughness, working with low frequencies minimizes the interferences from vegetation.

2. Scanning and Experience with Scanning

A trade study appears justified to compare phased arrays against higher orbit, larger antennas with correspondingly narrower angles for a scan. A number of electrically scanned microwave radiometers have been flown to date. Two of the Nimbus series spacecraft have successfully flown scanning experiments with both frequencies and resolutions relatable to soil moisture radiometers. The forthcoming Nimbus G and Seasat A will have scanning multichannel microwave radiometers in the frequency range 6 through 37 GHz and resolutions in the range 20 to 125 KM^2.

38181

DESIGN CONSIDERATIONS FOR LARGE SPACE ANTENNAS

JANUARY 1978

R. JOHNSON, JR.

MCDONNELL DOUGLAS

196

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PRESENTATION AREAS

- PROJECTED GROWTH OF LARGE SPACE SYSTEMS
- TYPICAL ANTENNA STRUCTURAL SYSTEMS
- CONCEPTS AND PERFORMANCE
- REQUIRED RESEARCH AND DEVELOPMENT

Figure 1

197

198

FUTURE LARGE SPACE SYSTEMS (Figure 2)

ON-ORBIT ASSEMBLY OF A 125-m LARGE APERTURE TEST ANTENNA IS DEPICTED IN THIS CONCEPTUAL APPROACH TO A POTENTIAL USE OF LARGE SPACE SYSTEMS. KEY ELEMENTS FOR SUCH OPERATIONS WILL INCLUDE THE SHUTTLE, A POWER MODULE, AND ASSEMBLY EQUIPMENT SUCH AS A SPACE CRANE. A CONSTRUCTION MODULE IS ALSO SHOWN IN THIS CONCEPT.

ON-ORBIT ASSEMBLY OPERATIONS CONCEPT

Figure 2

199

200

PROJECTED GROWTH OF LARGE SPACE SYSTEMS (Figure 3)

THE GROWTH IN DEVELOPMENT AND OPERATIONAL USE OF LARGE SPACE SYSTEMS IS EXPECTED TO FOLLOW A LOGICAL SEQUENCE FROM INITIAL EXPERIMENTAL DEPLOYMENT AND ASSEMBLY OPERATIONS CONDUCTED WITH THE SHUTTLE TO INITIAL SYSTEM DEPLOYMENT WITH SHUTTLE SUPPORT AND FINALLY TO LARGER, MORE COMPLEX SYSTEMS DEPLOYED AND SUSTAINED THROUGH USE OF PERMANENT SPACE CONSTRUCTION BASES. LATER DEPLOYMENT OF VERY LARGE SYSTEMS WILL REQUIRE AN ADVANCED LAUNCH VEHICLE.

PROJECTED OPERATIONAL GROWTH IN LARGE SPACE SYSTEMS

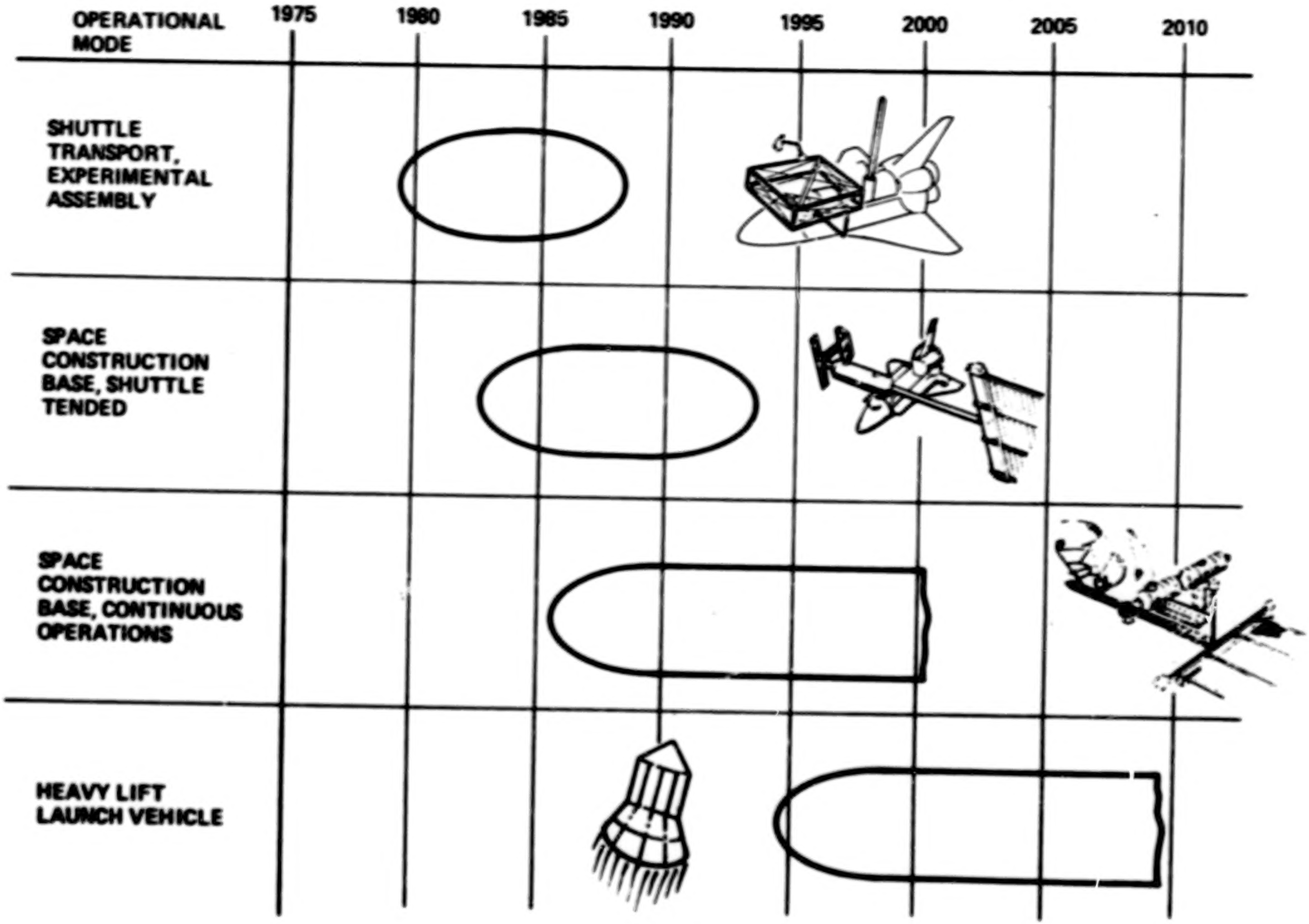


Figure 3

201

GROWTH OF WORLD COMMUNICATIONS (Figure 4)

AN EXAMPLE OF THE GROWTH PROJECTED IN ONE AREA WHERE LARGE SPACE SYSTEMS CAN BE APPLIED IS FOUND IN WORLD COMMUNICATIONS. THIS CHART SHOWS PAST AND PROJECTED GROWTH IN OVERSEAS TELEPHONE CALLS AS WELL AS THE DECREASING COST OF CALLS. FUTURE USE OF LARGE SPACE SYSTEMS CAN ASSIST SUBSTANTIAL GROWTH RATES WHILE MAINTAINING LOW COSTS.

GROWTH OF WORLD COMMUNICATIONS

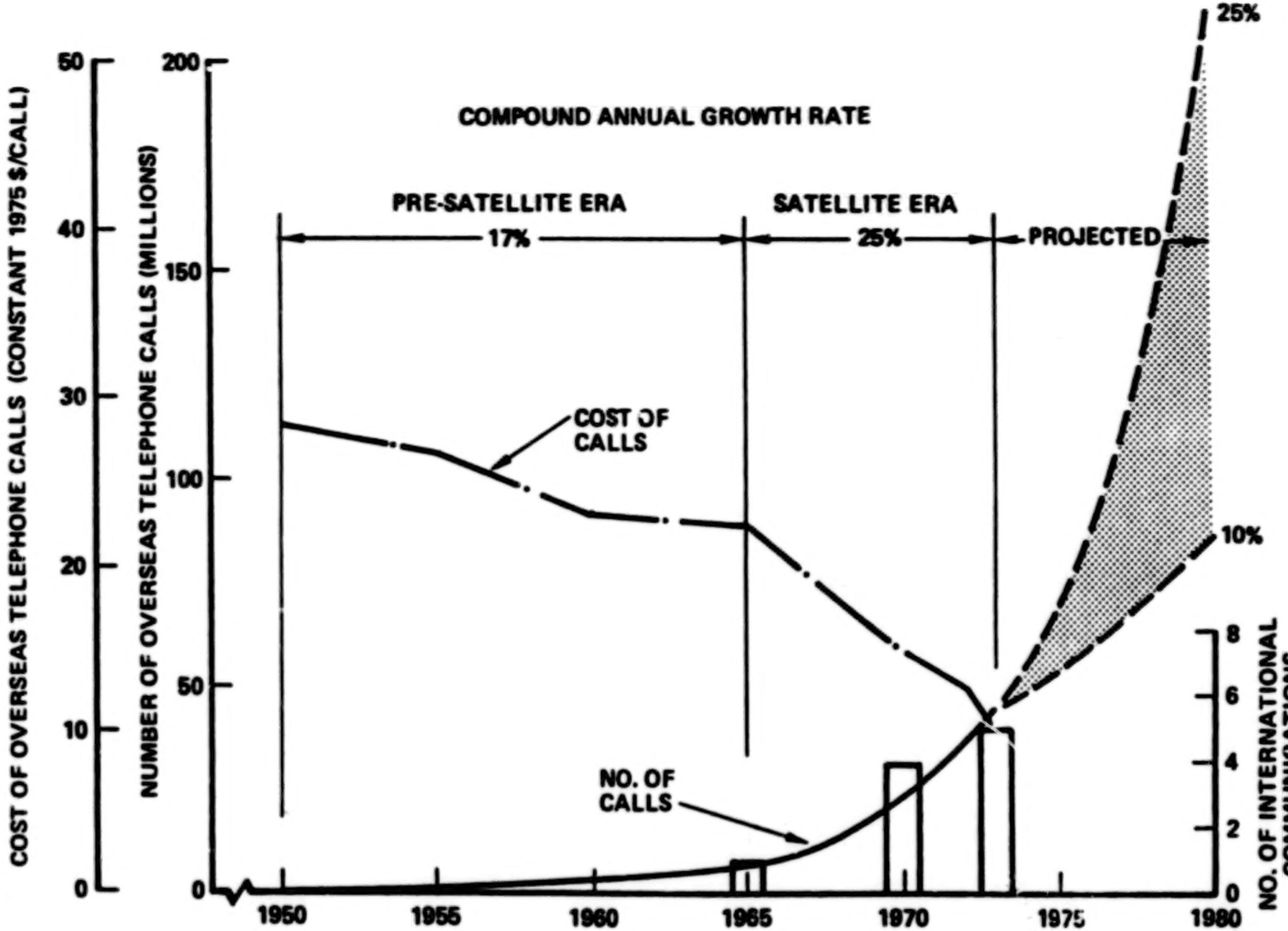


Figure 4

SOURCE: U.S. STATISTICAL ABSTRACT

203

TYPICAL LARGE SPACE STRUCTURES (Figure 5)

LARGE SPACE ANTENNAS WILL RANGE IN SIZE FROM APPROXIMATELY 30 m IN DIAMETER TO 1 km IN DIAMETER. ANTENNA TYPES WILL INCLUDE RADIOMETERS, MULTIBEAM LENS ANTENNAS, AND MICROWAVE POWER TRANSMISSION SYSTEMS.

TYPICAL LARGE SPACE STRUCTURES

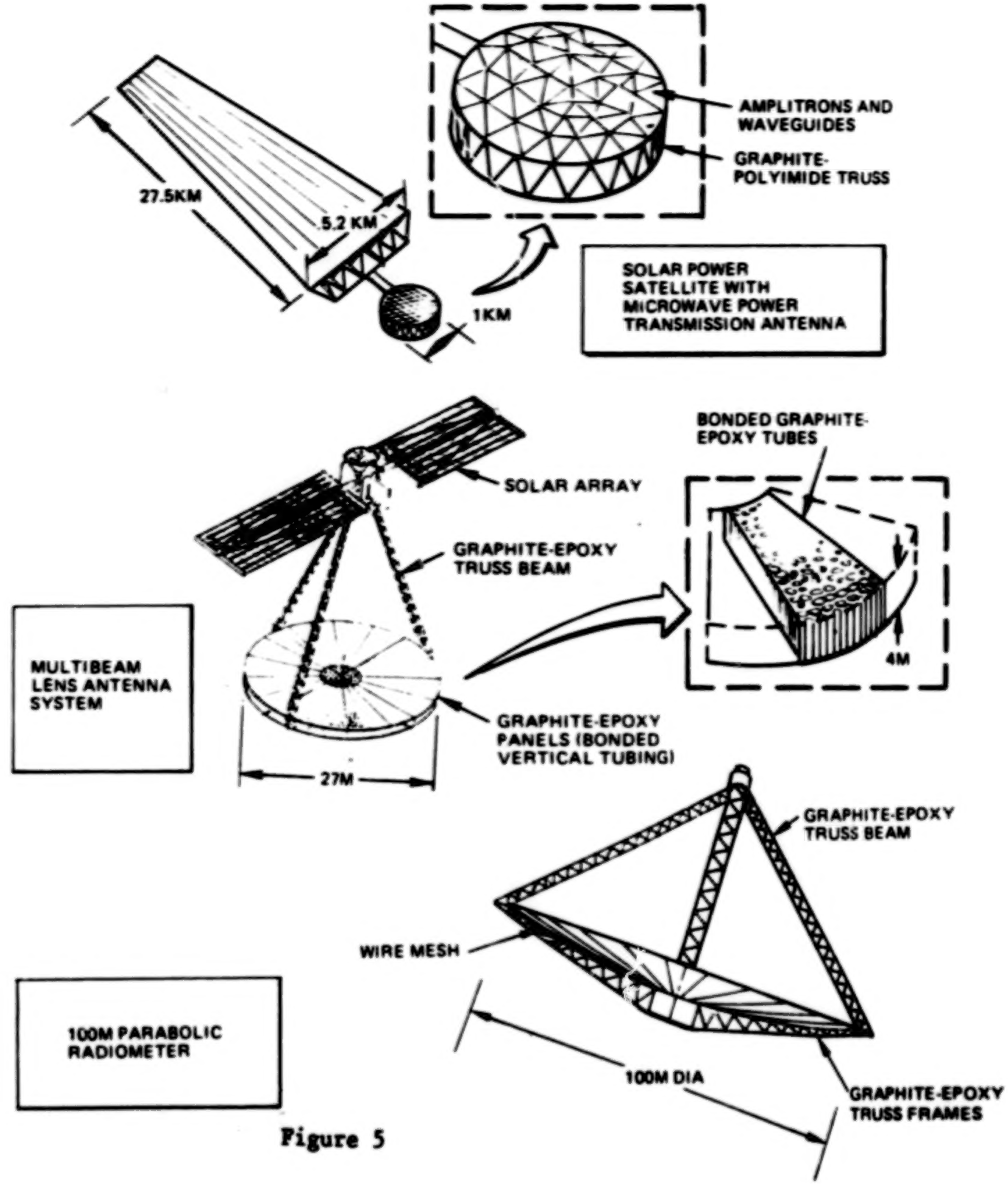


Figure 5

206

PERFORMANCE AND COST (Figure 6)

KEY PERFORMANCE AND COST FACTORS WILL PROVIDE SIGNIFICANT DESIGN DRIVERS IN THE DEVELOPMENT OF LARGE ANTENNAS. MINIMUM DISTORTION WILL BE A PRIMARY PERFORMANCE FACTOR AND COST DRIVERS WILL INCLUDE TRANSPORTATION CONSIDERATIONS PLUS ON-ORBIT FABRICATION AND ASSEMBLY TECHNIQUES DEVELOPED FOR LARGE SYSTEMS.

REQUIRED PERFORMANCE FOR MINIMUM COST

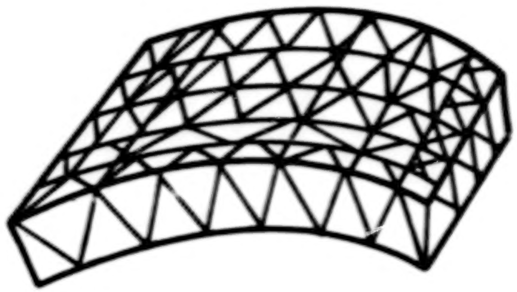
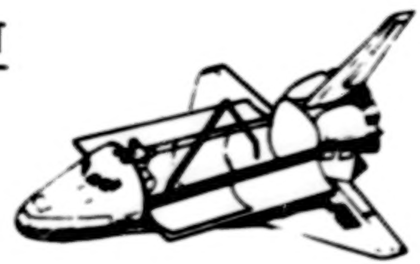
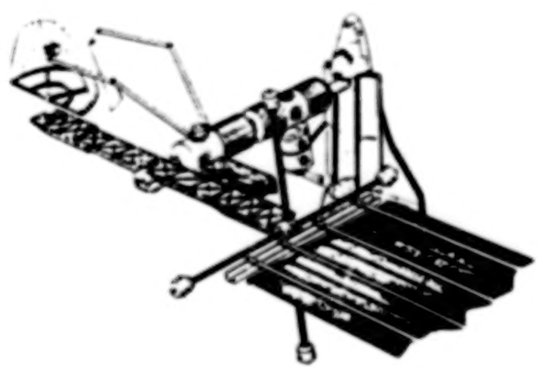
PERFORMANCE	DESIGN DRIVERS	KEY PARAMETERS
	• MINIMUM DISTORTION	• SPECIFIC MODULUS • COEFFICIENT OF THERMAL EXPANSION
COST 	• TRANSPORTATION	• WEIGHT • PACKAGEABILITY
	• FABRICATION AND ASSEMBLY	• SIMPLICITY • COMMONALITY

Figure 6

208

THERMAL DISTORTIONS (Figure 7)

ANTENNA ANGULAR DISTORTIONS ARE SHOWN HERE AS A FUNCTION OF COLUMN LENGTH FOR A TETRAHEDRAL TRUSS STRUCTURE. THE DISTINCT ADVANTAGE OF ADVANCED COMPOSITES IN REDUCING STRUCTURAL DISTORTION TO ACCEPTABLE LEVELS IS EVIDENT.

THERMAL DISTORTION COMPARISONS

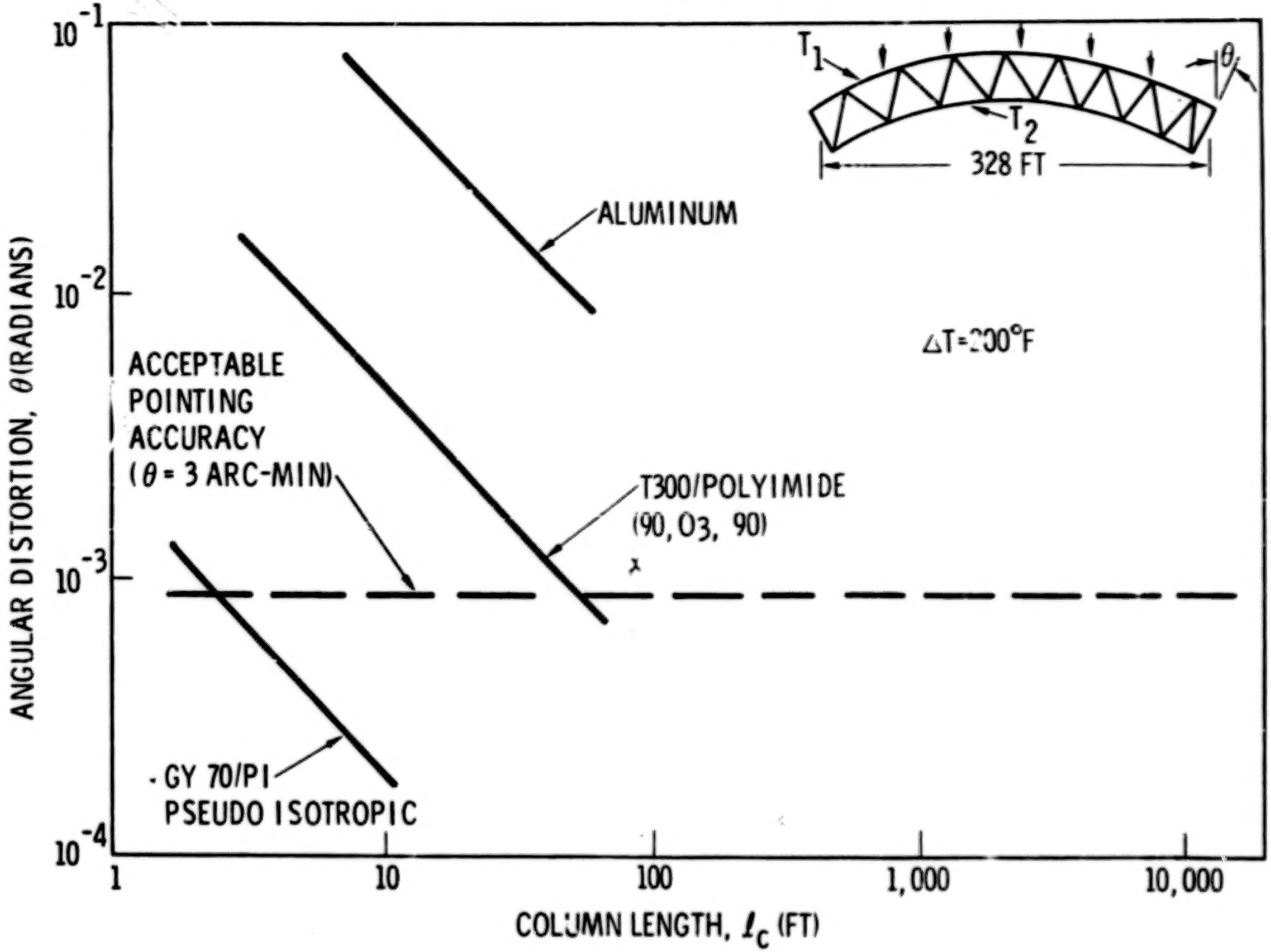


Figure 7

209

NESTABLE COLUMN WEIGHTS (Figure 8)

TWO TYPES OF NESTABLE COLUMNS WERE COMPARED TO DETERMINE WEIGHT DIFFERENCES AND CONSTRAINTS IMPOSED BY THE SHUTTLE PAYLOAD LIMITS IN TRANSPORTING THE STRUCTURAL ELEMENTS TO LOW EARTH ORBIT. THE COLUMNS ARE TRANSPORTED IN TAPERED HALF-SEGMENTS NESTED IN LONGITUDINALLY ALIGNED GROUPS AND ASSEMBLED ON ORBIT BY JOINING PAIRS OF HALF SEGMENTS TO FORM FULL COLUMNS. FOR A GIVEN DESIGN LOAD THE OPEN TRUSS TYPE CONFIGURATION SHOWS A WEIGHT ADVANTAGE COMPARED TO THE CLOSED CIRCULAR DESIGN BEFORE PACKAGING CONSTRAINTS ARE CONSIDERED.

NESTABLE COLUMN WEIGHT COMPARISONS

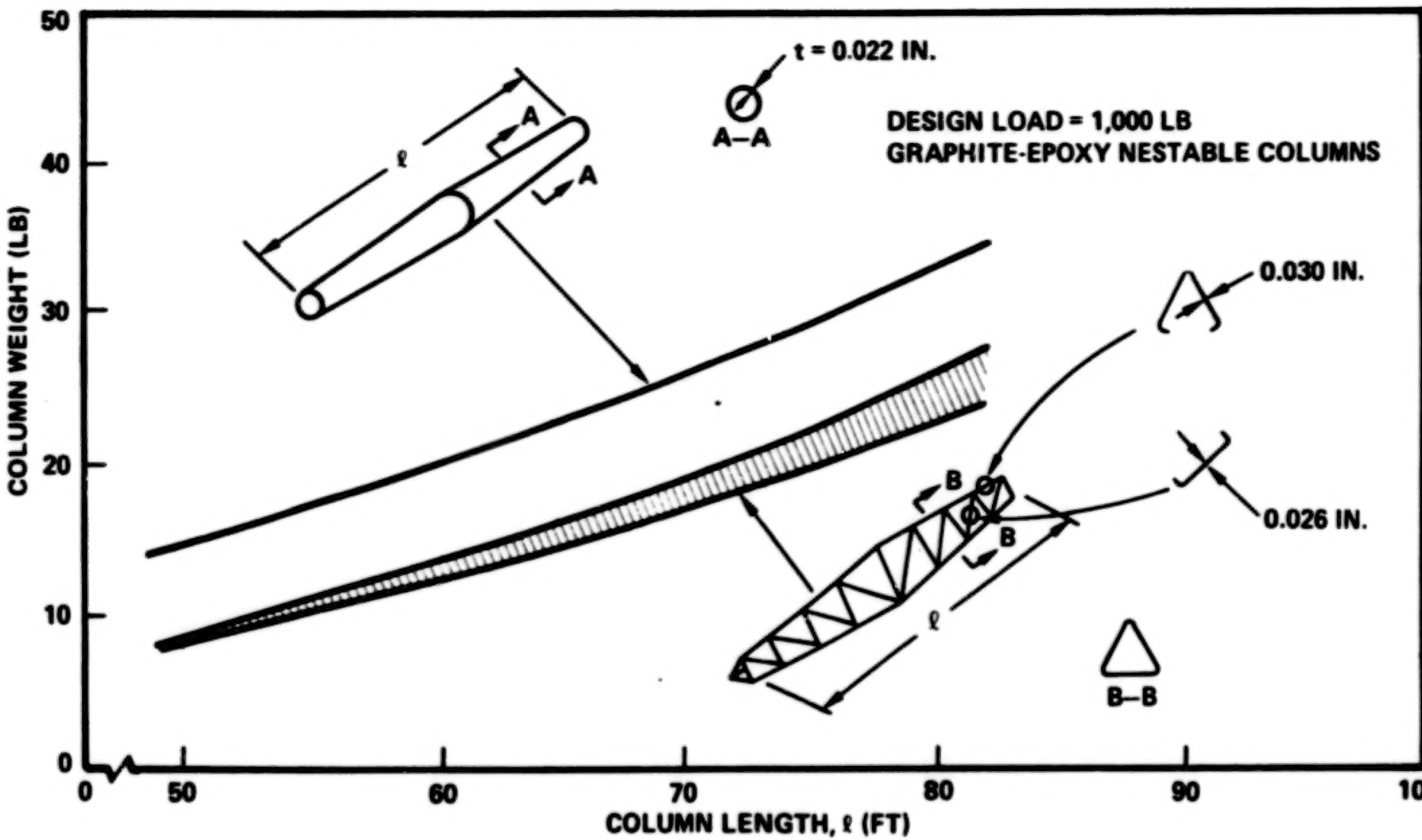


Figure 8

212

NUMBER OF NESTABLE COLUMNS PER SHUTTLE FLIGHT (Figure 9)

A GREATER NUMBER OF OPEN-TRUSS TYPE NESTABLE COLUMNS CAN BE PACKAGED IN THE SHUTTLE WHEN CONSIDERING PAYLOAD LIMITS ONLY. HOWEVER, SHUTTLE CENTER OF GRAVITY CONSTRAINTS CAN HAVE A SIGNIFICANT EFFECT, AS SHOWN IN THE FOLLOWING CHART.

NUMBER OF NESTABLE COLUMNS PER SHUTTLE FLIGHT

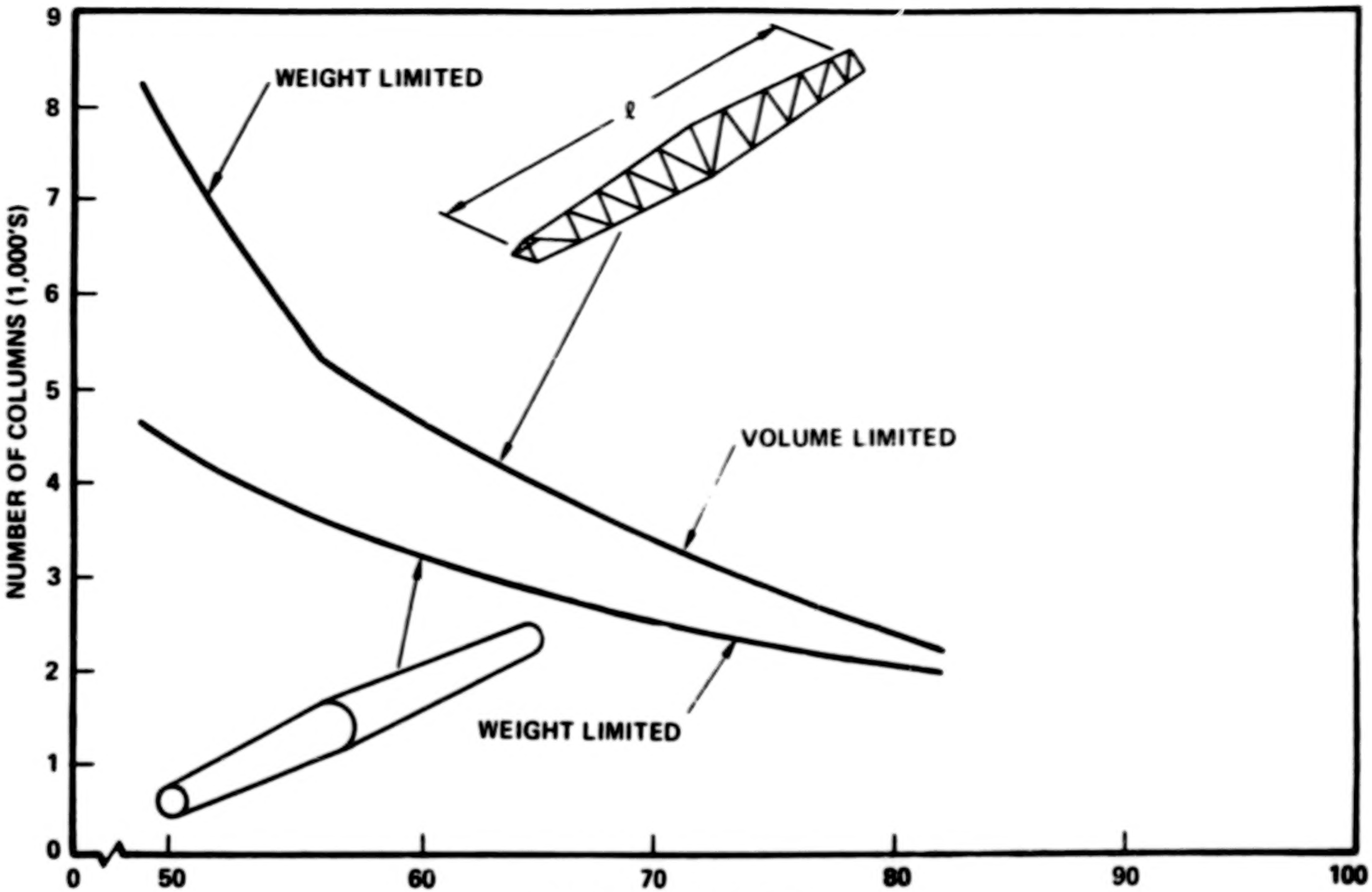


Figure 9

213

EFFECT OF SHUTTLE CENTER OF GRAVITY CONSTRAINTS (Figure 10)

THE EFFECT OF SHUTTLE CG CONSTRAINTS IS TO REDUCE THE NUMBER OF OPEN TRUSS COLUMNS AVAILABLE PER SHUTTLE FLIGHT DUE TO THAT CONCEPT'S LOWER PACKAGING DENSITY. THE HIGHER PACKAGING DENSITY OF THE THIN WALLED CIRCULAR CLOSED COLUMN MAKES IT COMPETITIVE WITH THE OPEN TRUSS DESIGN OVER A MAJOR RANGE OF COLUMN LENGTHS OF INTEREST. THE SIMPLER DESIGN OF THE CLOSED CIRCULAR COLUMN CAN ALSO PROVIDE A LOWER FABRICATION COST WHEN COMPARED TO THE OPEN TRUSS DESIGN.

NUMBER OF NESTABLE COLUMNS PER SHUTTLE FLIGHT WITH CG CONSTRAINTS

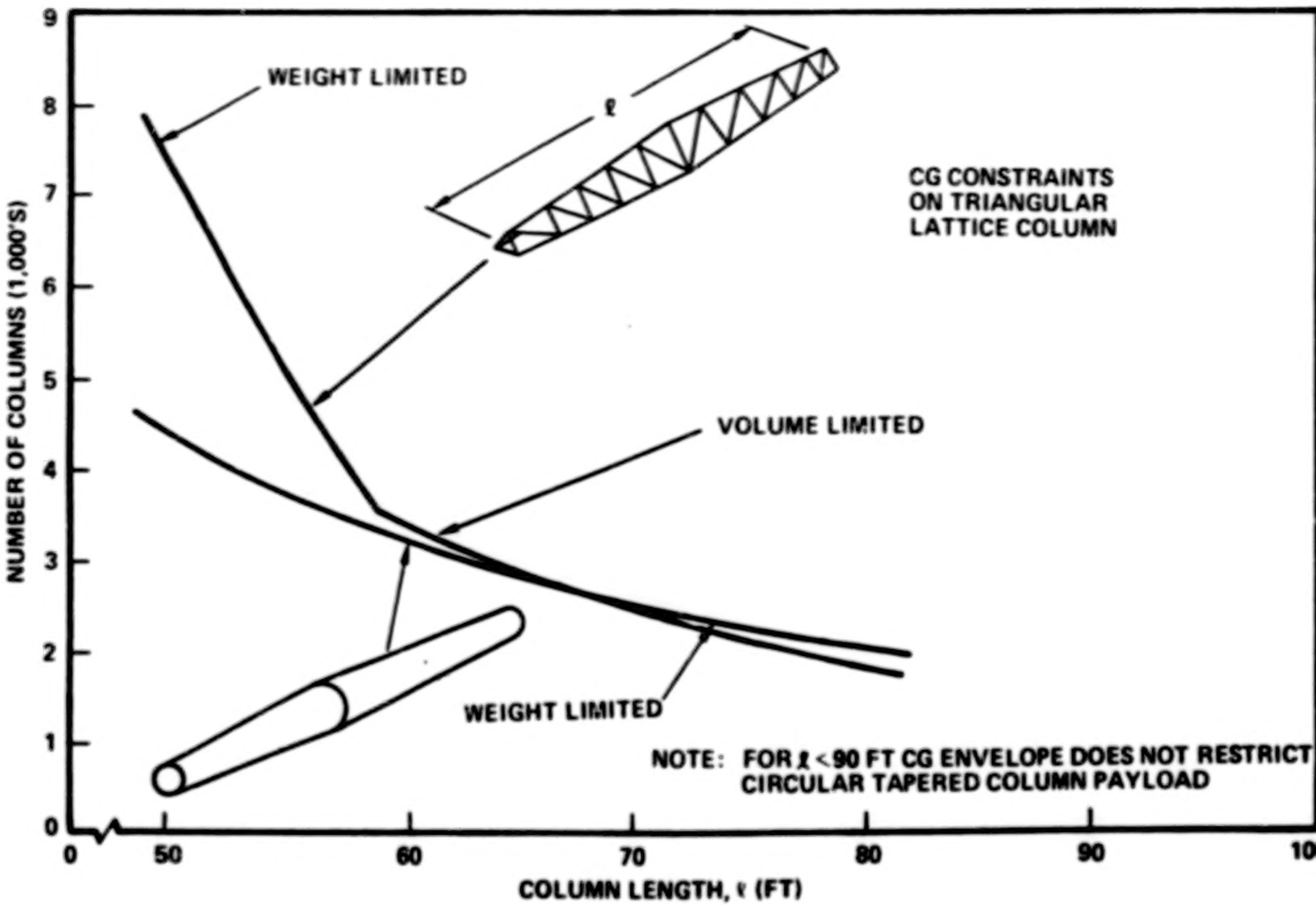


Figure 10

215

216

JOINT WEIGHTS (Figure 11)

THE INCREASED STRUCTURAL MASS DUE TO JOINT WEIGHTS IS SHOWN HERE FOR A TETRAHEDRAL TRUSS ANTENNA STRUCTURE. THE GREATER PERCENTAGE OF JOINT WEIGHT FOR SHORTER COLUMNS IS EVIDENT IN THE ACCOMPANYING PLOT OF ANTENNA MASS VERSUS VOLUMN LENGTH.

ANTENNA STRUCTURAL WEIGHT

(100M SQUARE ANTENNA CONSTRUCTED OF GY70/PI)

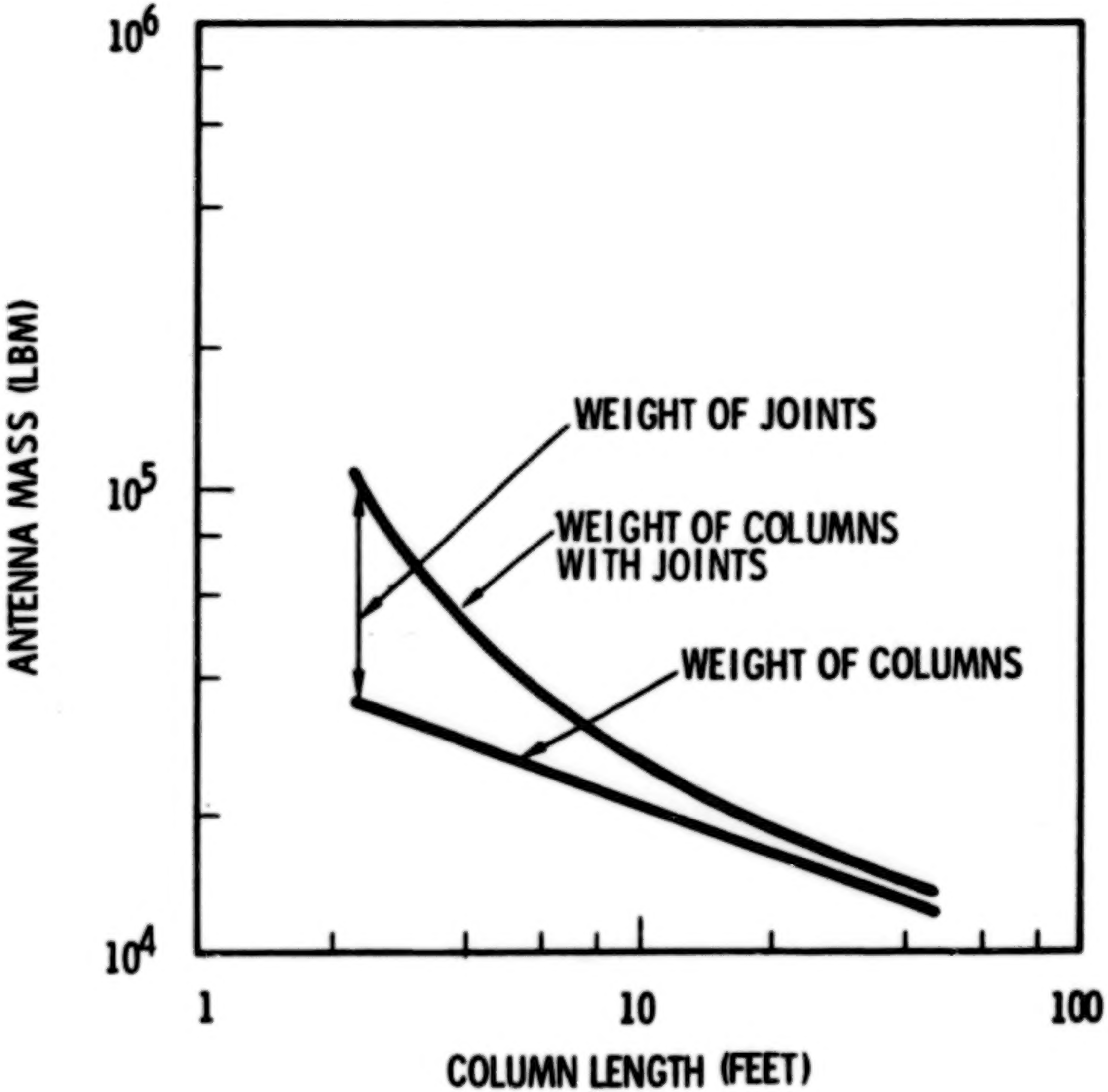


Figure 11

217

REQUIRED RESEARCH AND DEVELOPMENT (Figure 12)

REQUIRED RESEARCH AND DEVELOPMENT AREAS FOR LARGE SPACE STRUCTURES ARE SUMMARIZED IN THIS CHART. DEVELOPMENT EFFORTS ARE REQUIRED IN A NUMBER OF AREAS AND BOTH GROUND TESTING AND ORBITAL TESTING WILL BE NECESSARY ELEMENTS IN A SUCCESSFUL DEVELOPMENT PROGRAM.

REQUIRED STRUCTURES RESEARCH AND DEVELOPMENT

- DEFINE COMPREHENSIVE STRUCTURAL CRITERIA FOR SPECIFIC MISSIONS
- DETERMINE OPTIMIZED GEOMETRIES FOR TRUSS ELEMENTS AND TRUSS CONFIGURATIONS
- ESTABLISH REQUIREMENTS FOR JOINT DESIGNS
  - TOLERANCES
  - LOAD TRANSFER
  - THERMAL COMPATIBILITY
  - ASSEMBLY/DISASSEMBLY
- DEVELOP CANDIDATE STRUCTURAL ELEMENTS AND JOINT CONCEPTS FOR STRUCTURES TO BE CONSTRUCTED BY DEPLOYMENT, ASSEMBLY ONLY, AND ON-ORBIT FABRICATION AND ASSEMBLY TECHNIQUES
- DEVELOP MATERIAL SYSTEMS FOR LOW PRESSURE, RAPID CURE OF ADVANCED COMPOSITE MATERIALS
- DEVELOP VARIOUS ON-ORBIT ASSEMBLY AND FABRICATION TECHNIQUES

Figure 12

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LARGE ANTENNA STRUCTURE TECHNOLOGIES REQUIRED FOR 1985 — 2000

W. R. WANNLUND

TRW

ONE SPACE PARK • REDONDO BEACH • CALIFORNIA, 90278

222

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January 9, 1978

LARGE ANTENNA STRUCTURE TECHNOLOGIES

REQUIRED FOR 1985 - 2000

Paper for presentation to the Joint Government/Industry Seminar on Large Space Systems Technology to be held January 17-19, 1978 at Langley Research Center, VA.

ABSTRACT

In order to focus on the future technology requirements for large antennas, TRW DSSG is postulating a typical hardware (mission) end item and forecasting the technology needs to support that end item. The subjects discussed include:

- Material degradation as related to graphite composites and thermal control coatings.
- Thermal distortions considering postulated end-of-life conditions for extended lifetime (over 10 years).
- Examination of built-in shape versus actively controlled surfaces.
- Testing philosophy of super large antenna structures.
- Examination of some "what-if's" which may require new or different technology.

224

<u>Configuration Selection</u>: (Figure 1)

- 20-year life communication antenna - revenue required to offset cost of system.
- Deployable center section and an assembled rib mesh - covers two major technology areas.
- Materials used - graphite-epoxy face sheeted aluminum or Kevlar core honeycomb, graphite-epoxy ribs, appropriate mesh.

<u>Introduction</u>:

A large number of "technology needs" are satisfied in the execution of an actual hardware contract and are considered part of the normal work. Some of the items that I would include in this category are:

- Repair and refurbishment techniques.
- Contour measuring and methods/devices.
- Shuttle environments and constraints - loads, contamination, thermal, etc.
- Deployment mechanisms, damping devices, drives.
- Assembly in space techniques.
- Joint design, element design, subsystem and system design.

For the purpose of this paper I have tried to focus on a few items which ought to be solved before starting a hardware contract. It turned out that most of the requirements came from the selection of nonmetallic components. In addition, with the few minutes available, I felt it would be worthwhile sharing a little of our philosophy in the analysis and test area.

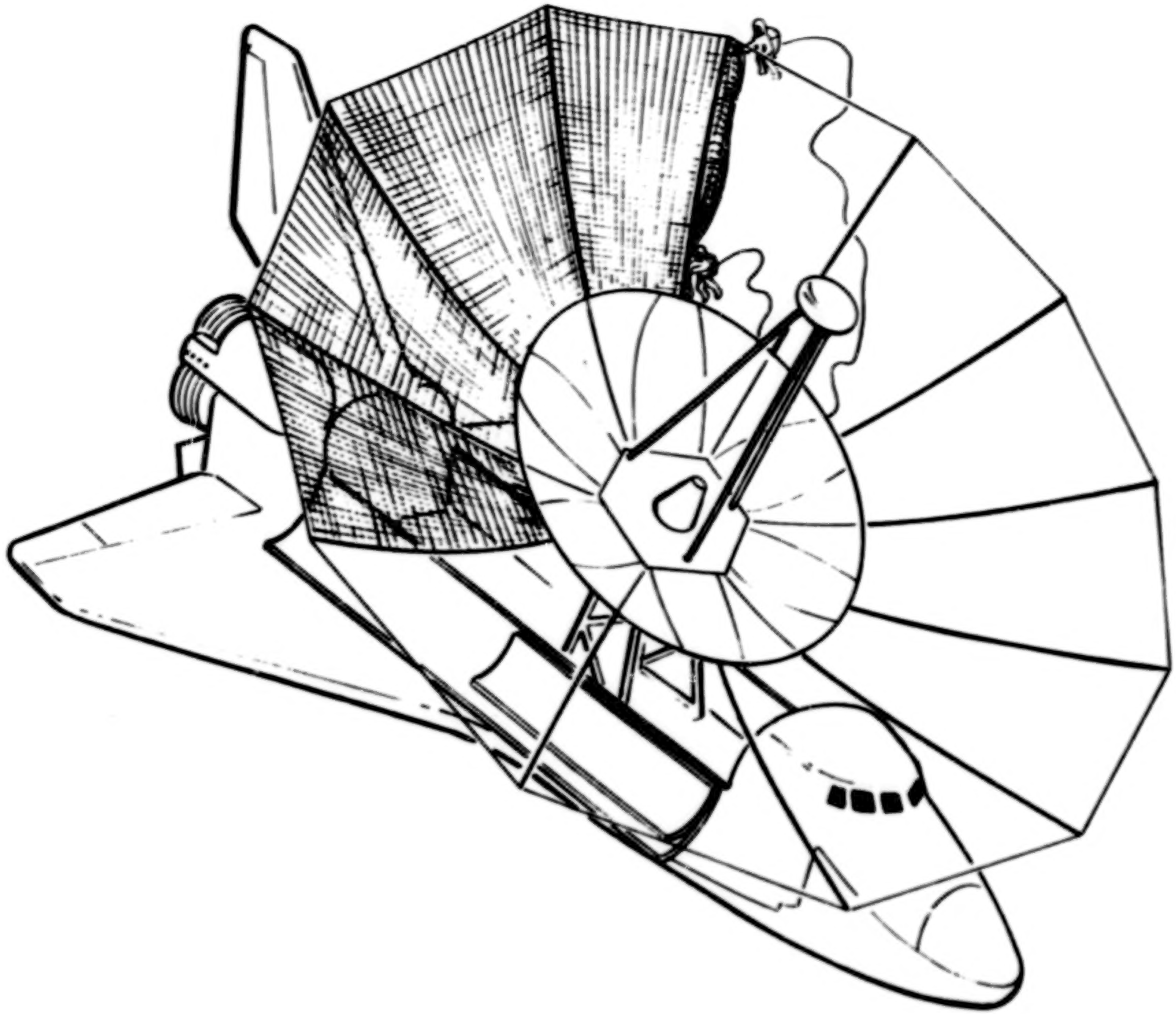


FIGURE 1

225

226

I believe some key technology problems are generated because of extended lifetimes of 10 to 20 years. It is highly probable that very large systems will have to remain in service in excess of 10 years to achieve economic realism. Technologies required will include repair and refurbishment.

Pertinent Technologies:

Material degradation in space environment: (Figure 2)

1. Thermal control coatings can be assumed to degrade to an unacceptable range in 10 to 20 years in space environment.

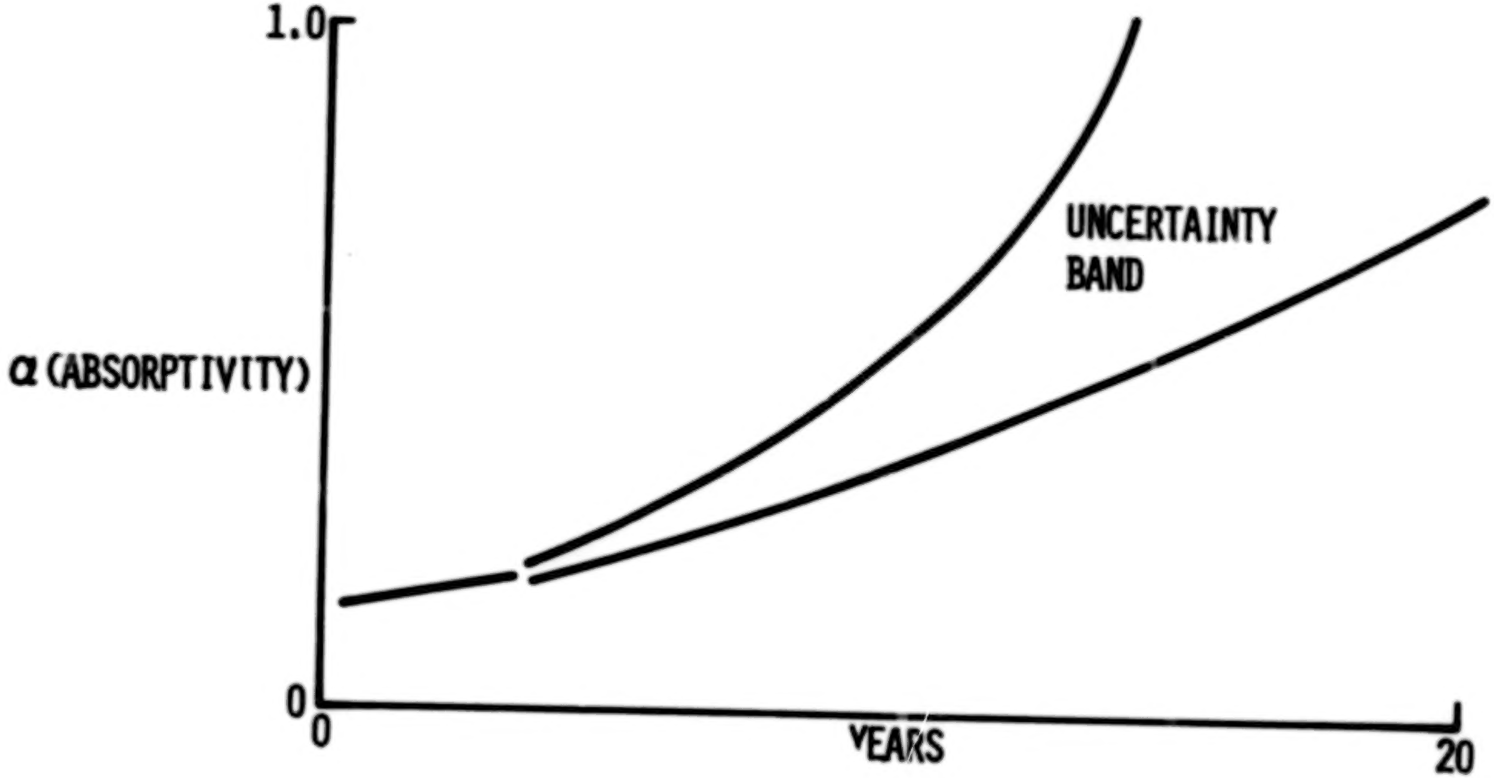
   Resultant graphite-epoxy rib temperatures would approach +250°F in Sun and -320°F in eclipse.

Problem:

Most attachments are secondarily bonded to the rib with room cure adhesives (applied at mid range of thermal cycles). Needed are room cure adhesives which provide sufficient structural margins to accommodate maneuvering loads at extreme conditions, both hot and cold.

Note:

Thermal distortion is covered later.



THERMAL COATINGS

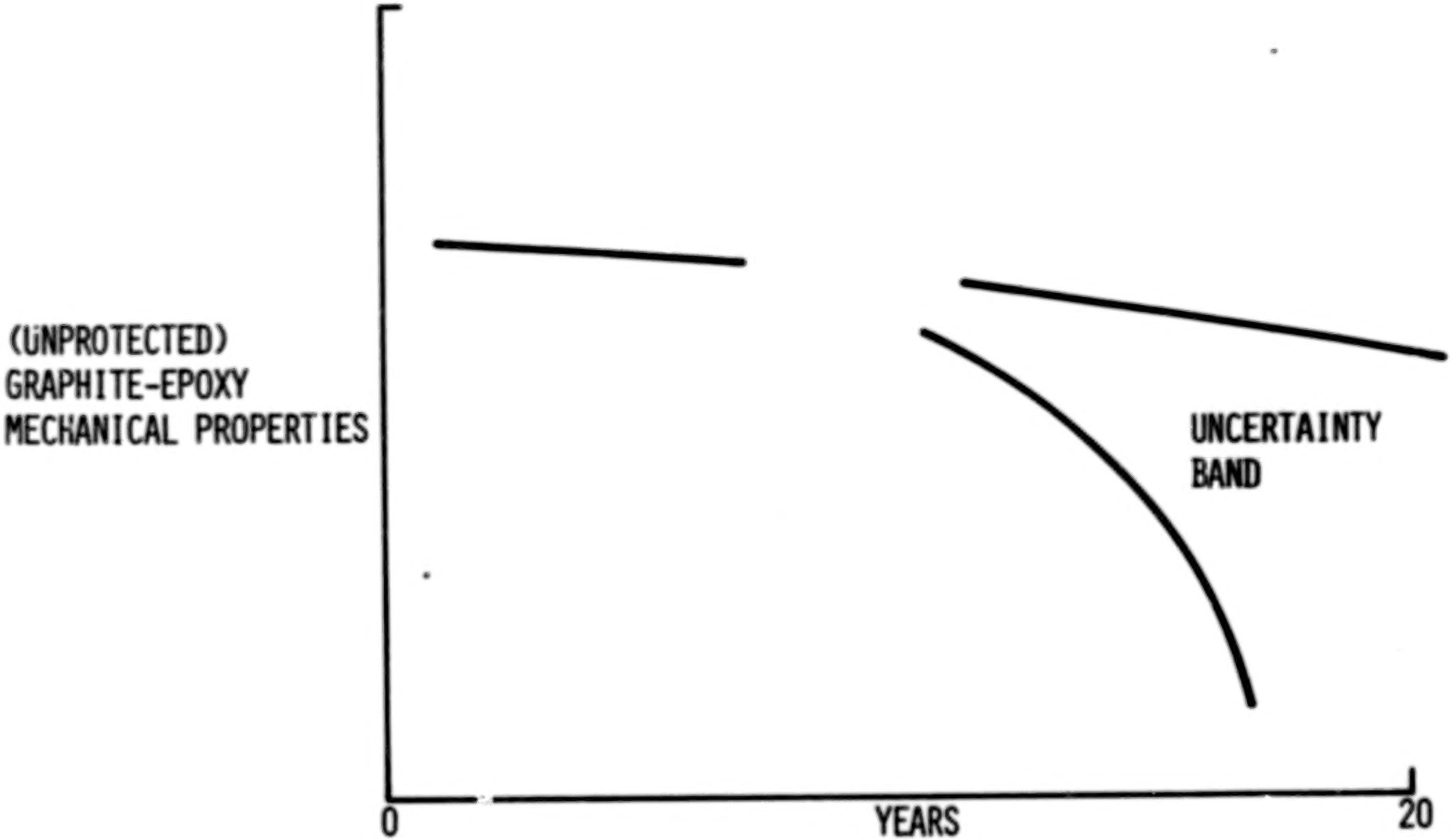
FIGURE 2

227

2. Graphite-Epoxy Degradation (Figure 3)

Organic resin systems should conservatively be considered to degrade excessively in the 10 to 20 year span due to ultra violet and charged particle exposure in space.

This points up the need for the development of an acceptable barrier coating or the development of impervious synthetic resin systems. In our aerospace culture it is not sufficient to state that present materials are not adequate for some missions; it is necessary to show by measurement at what point the materials become inadequate. This brings up a technology I would like to emphasize.



GRAPHITE-EPOXY MECHANICAL PROPERTIES

FIGURE 3

CONTENTS

VOLUME I

230

Measurement Facilities and Technology

At TRW we set up and operated a special test facility to gather materials properties data on various material samples while being exposed to accelerated dosages of ultra-violet, high-energy electrons, low-energy electrons, and protons.

Figure 4 shows the time necessary to gather extended life data in our facility as presently operated.

CHAMBER (TRW)	U.V. RATE	TEST TIME (YRS)	
		SIMULATED 10 YRS	SIMULATED 20 YRS
IN-SITU OPTICAL	3.3 π	1	2
EX-SITU MECHANICAL	2.5 π	1.25	2.5
IN-SITU TENSILE	1.5 π	2.1	4.2

TIME TO ACQUIRE EXTENDED LIFE DATA
FIGURE 4

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231

232

The facility layout is shown in Figure 5A and Figure 5B and shows the complexity of the equipment required.

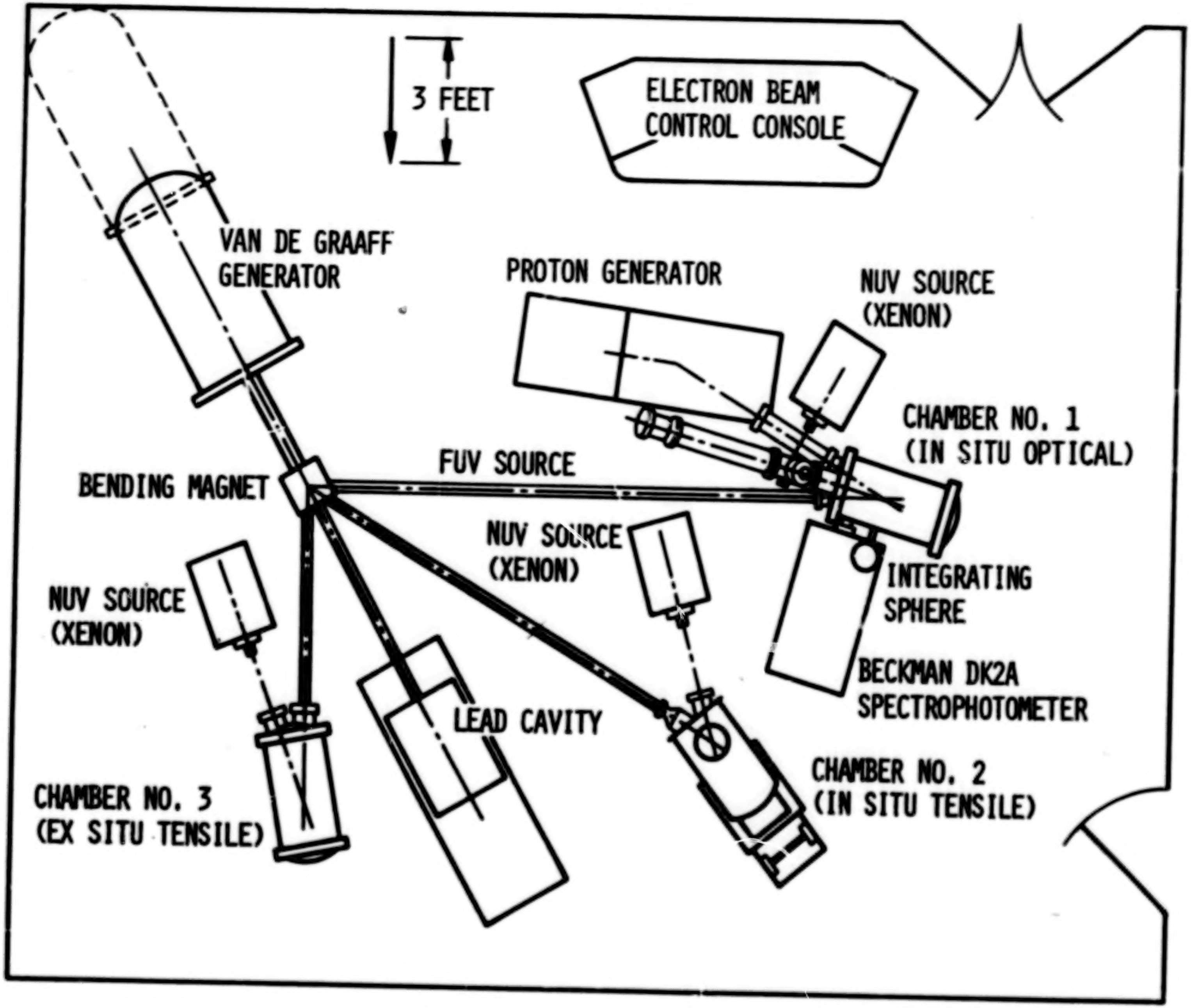

FIGURE 5A LABORATORY LAYOUT

Problem:

A need exists to develop experimentally the effects of accelerated dosages of U.V. and charged particles on the mechanical and optical properties of graphite-epoxies and coatings and other plastic materials. When one accelerates the dosage rates beyond what we have done, the resultant damage cannot be correlated.

For example: The heat content of a U.V. source which has the correct spectral quality when operating at several suns may generate very unrealistic and excessive damage to the sample.

The whole materials technology area related to accelerated exposures in both facilities aspects and damage mechanisms needs attention.

Note:

Four years continuous operation to get graphite-epoxy tensile data to 20 years is not very practical.

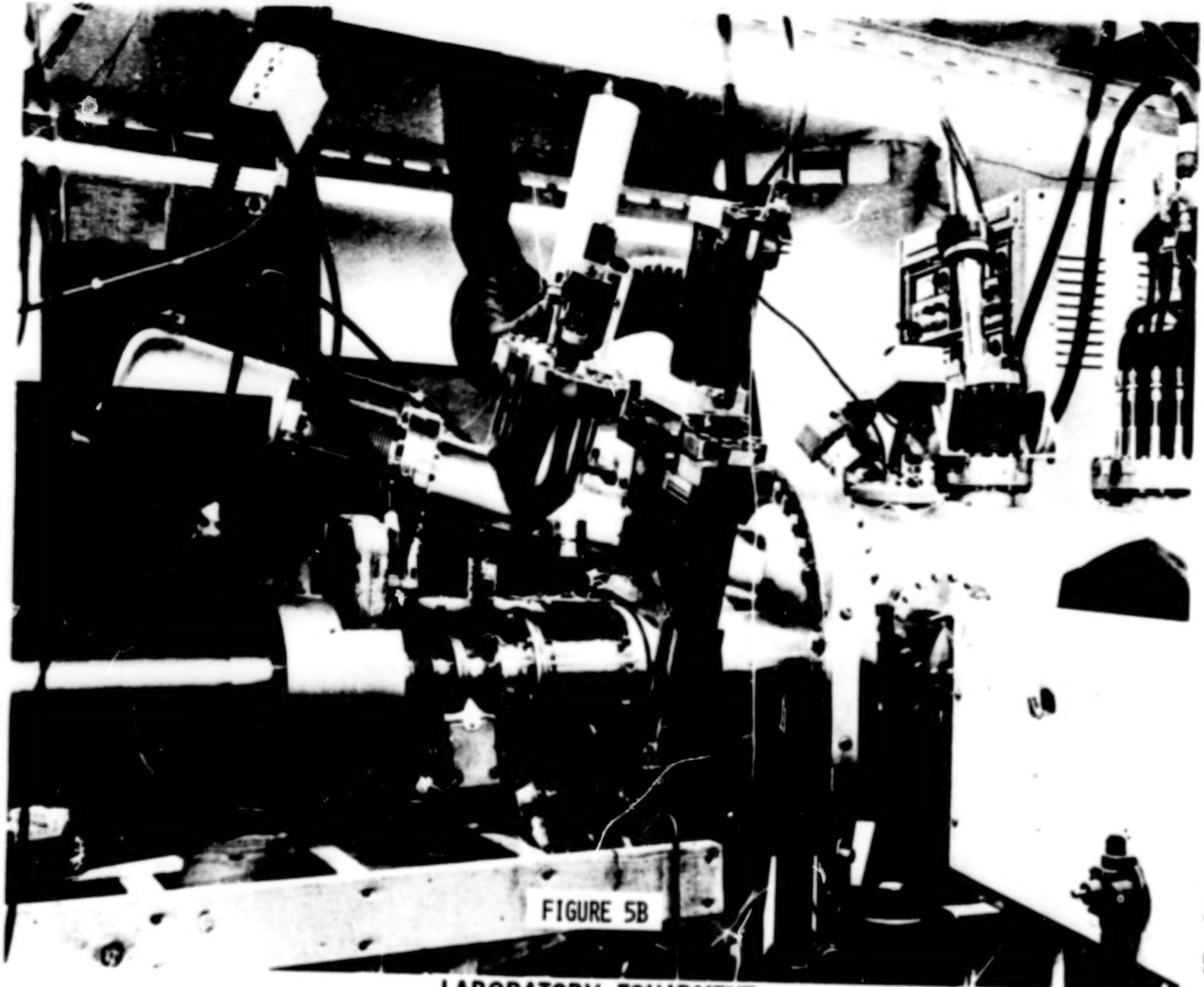

FIGURE 5B

LABORATORY EQUIPMENT

236

Thermal Distortion - Built-in Shape Versus Active Controls (Figure 6)

Degradation (with effective U.V. barrier coating) of coefficient of thermal expansion is predicted to be in the same order as those differences due to process variation, manufacturing tolerances, and other secondary effects.

Conclusion:

Modest size graphite-epoxy ribbed mesh antenna reflectors for 6 GHz and under (150' dia.) can keep within satisfactory performance range without active controls; very large reflectors will need some type of adjustment feature or active controls such as movable feeds, subreflectors or rigging.

Note:

Without barrier coatings based on today's knowledge, one would be forced into a cost trade-off with metallic ribs and active controls to meet 20-year performance criteria.

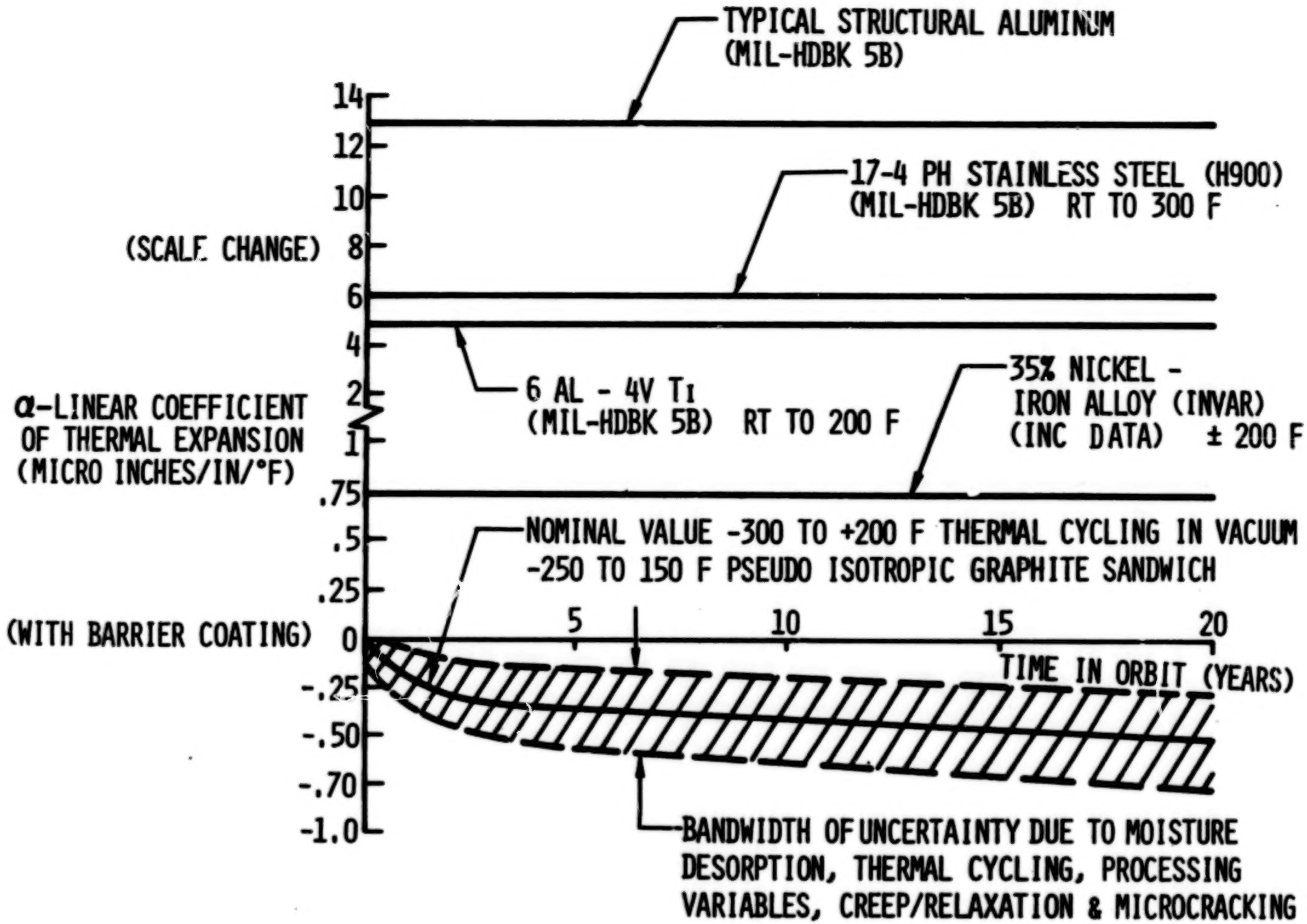
TYPICAL STRUCTURAL ALUMINUM
(MIL-HDBK 5B)
17-4 PH STAINLESS STEEL (H900)
(MIL-HDBK 5B) RT TO 300 F
(SCALE CHANGE)
6 AL - 4V Ti
(MIL-HDBK 5B) RT TO 200 F
35% NICKEL -
IRON ALLOY (INVAR)
(INC DATA) ± 200 F
α-LINEAR COEFFICIENT
OF THERMAL EXPANSION
(MICRO INCHES/IN/°F)
NOMINAL VALUE -300 TO +200 F THERMAL CYCLING IN VACUUM
-250 TO 150 F PSEUDO ISOTROPIC GRAPHITE SANDWICH
(WITH BARRIER COATING)
TIME IN ORBIT (YEARS)
BANDWIDTH OF UNCERTAINTY DUE TO MOISTURE
DESORPTION, THERMAL CYCLING, PROCESSING
VARIABLES, CREEP/RELAXATION & MICROCRACKING
14
12
10
8
6
4
2
1
.75
.5
.25
0
-.25
-.50
-.70
-1.0
5
10
15
20

FIGURE 6

237

238

Testing Philosophy - Super Large Antenna Structures (Figure 7 and Figure 8)

Because of the impracticalities of testing, large rib mesh reflector systems rely heavily on analytical models to determine on-orbit contour. Mechanical tolerances (including manufacturing, processes, materials properties, analytical approximations, and measurements) result in an uncertainty in the predicted contour which may be of first order magnitude. Our philosophy is to develop "serialized analytical models" for the required on-orbit contour predictions.

A serialized analytical model is a simulation which uses mechanical properties based on actual test of individual components of the structure. The simulation is not synthesized by using average material properties to predict behavior of individual structure components, as is the standard practice. The theoretical nature of the analytical model is unchanged, only member cross-section properties are based on actual member tests.

This approach results in analytical models which are high precision but are unique to the specific structural assembly being simulated. Mechanical tolerances are minimized or eliminated to the extent that they are included in the individual member tests. Symmetrical structures, however, do not result in symmetrical models because of differences in the structural properties of supposedly identical members. Not only is the structural assembly serialized but individual elements of each assembly are also uniquely serialized.

BASIC TEST APPROACH - LARGE RIB-MESH

- o PLACE EACH RIB INDIVIDUALLY IN A SIMULATED ZERO G CONDITION
- o APPLY SIMULATED THERMAL/STRUCTURAL LOADS
- o MEASURE DISPLACEMENTS (INFLUENCE COEFFIENTS)
- o CHECK ANALYTICAL MODEL - TUNE TO ACTUALS
- o COMPILE SERIALIZED ANALYTICAL MODEL
- o DESIRABLE: ASSEMBLE SEVERAL GORES IN SIMULATED ZERO G ($\top+\perp$) AND MEASURE RIB AND MESH DISPLACEMENTS

239

8710.0-4

FIGURE 7

" WHAT IF " IMPACTS ON LARGE ANTENNA STRUCTURES TECHNOLOGY

- PERTINENT RESINS GET OUTLAWED BY HEALTH AND SAFETY INTERESTS
  - IMPACT
    - METALLIC RIBS AND ACTIVE CONTROLS
    - TOTAL AUTOMATION IF RESIN ALLOWED TO BE PRODUCED BUT NOT HANDLED
    - DEVELOPMENT OF SYNTHETIC OR OTHER RESIN FORMULATIONS
- SHUTTLE FUNDS GOT "SST'D" BY CONGRESS
  - IMPACT
    - STAY WITH 2000 LB CLASS ANTENNA FARMS
    - CONSIDER "ALL COMMERCIAL" BOOSTER/SHUTTLE (INTERNATIONAL?)
- COLLISION, ACCIDENT, DAMAGE FROM ANY SOURCE
  - IMPACT
    - MANUAL REPAIR TECHNOLOGY FOR RIBS, MESH PATCHES INCLUDING CURING OF ADHESIVES IN SPACE OR TOTALLY MECHANICAL REPAIRS
- SOMEONE DEVELOPS A GOOD U.V. AND CHARGED PARTICLE BARRIER COATING FOR GRAPHITE EPOXIES WHICH ALSO SERVES AS A SUPER STABLE THERMAL CONTROL COATING BUT IT COST $5,000/SQ FT AS APPLIED
  - IMPACT:
    - NEED TO DEVELOP COST TRADE GUIDELINES FOR OUR INDUSTRY.

FIGURE 8

Summary

Basic materials work in the nonmetallics, including coatings, related to all physical properties for extended life times in space seems to be a high priority requirement. This includes facilities as well as some basic understanding of the physics of damage mechanisms during accelerated testing. Technologies for thermal coating systems under 10-year life seem within engineering extrapolation range - over 10 years appears beyond sound extrapolations.

Serialized analytical models appear to offer a reasonable compromise to the lack of ability to test some of the very large antenna structures.

241

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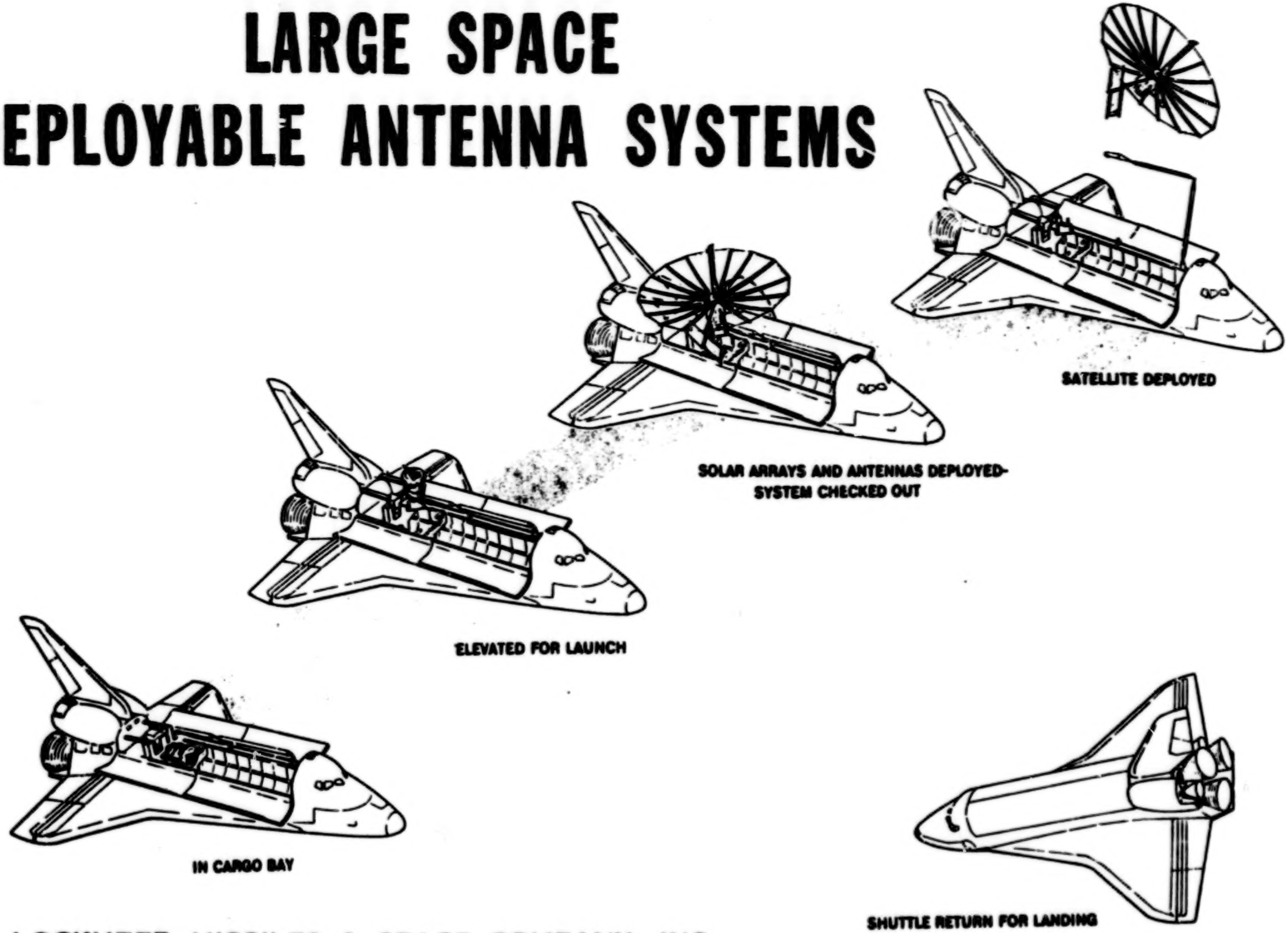
LARGE SPACE DEPLOYABLE ANTENNA SYSTEMS



LOCKHEED MISSILES & SPACE COMPANY. INC.
SPACE SYSTEMS DIVISION · SUNNYVALE, CALIFORNIA

G.G. CHADWICK, MANAGER RF/ANTENNA SYSTEMS
A.A WOODS JR, STAFF ENGINEER

244

(Figure 1)

SINCE ITS INCEPTION IN 1955, THE LOCKHEED MISSILES & SPACE COMPANY, INC., HAS BEEN ACTIVELY ENGAGED IN THE DESIGN AND DEVELOPMENT OF ADVANCED ANTENNA SYSTEMS FOR SPACE APPLICATIONS. OVER 300 DIFFERENT ANTENNA DESIGNS HAVE BEEN DEVELOPED, TESTED, AND FLOWN ON MANY DIFFERENT VEHICLES.

IN EARLY 1962, LMSC ANTENNA ENGINEERING CONCEIVED A PROPRIETARY DESIGN FOR A PARABOLOIDAL REFLECTOR. THIS CONCEPT, CALLED THE WRAP-RIB PARABOLA, WAS A BREAKTHROUGH IN THE SENSE THAT IT MADE POSSIBLE THE CONSTRUCTION OF ACCURATE SURFACES OF LARGE SIZE. AT THE TIME, SIZES UP TO 50 FT IN DIAMETER SEEMED POSSIBLE. SUBSEQUENT IMPROVEMENTS IN THE BASIC DESIGN APPROACH HAVE MADE THE INITIAL 50-FT FIGURE A MODEST ESTIMATE. REFLECTORS HAVE BEEN BUILT AT LMSC UP TO 30 FT IN DIAMETER DESIGNED FOR OPERATION AT X-BAND. THE LARGEST REFLECTOR THAT HAS BEEN PLACED ON ORBIT WAS THE X-BAND ATS 30-FT-DIAMETER REFLECTOR

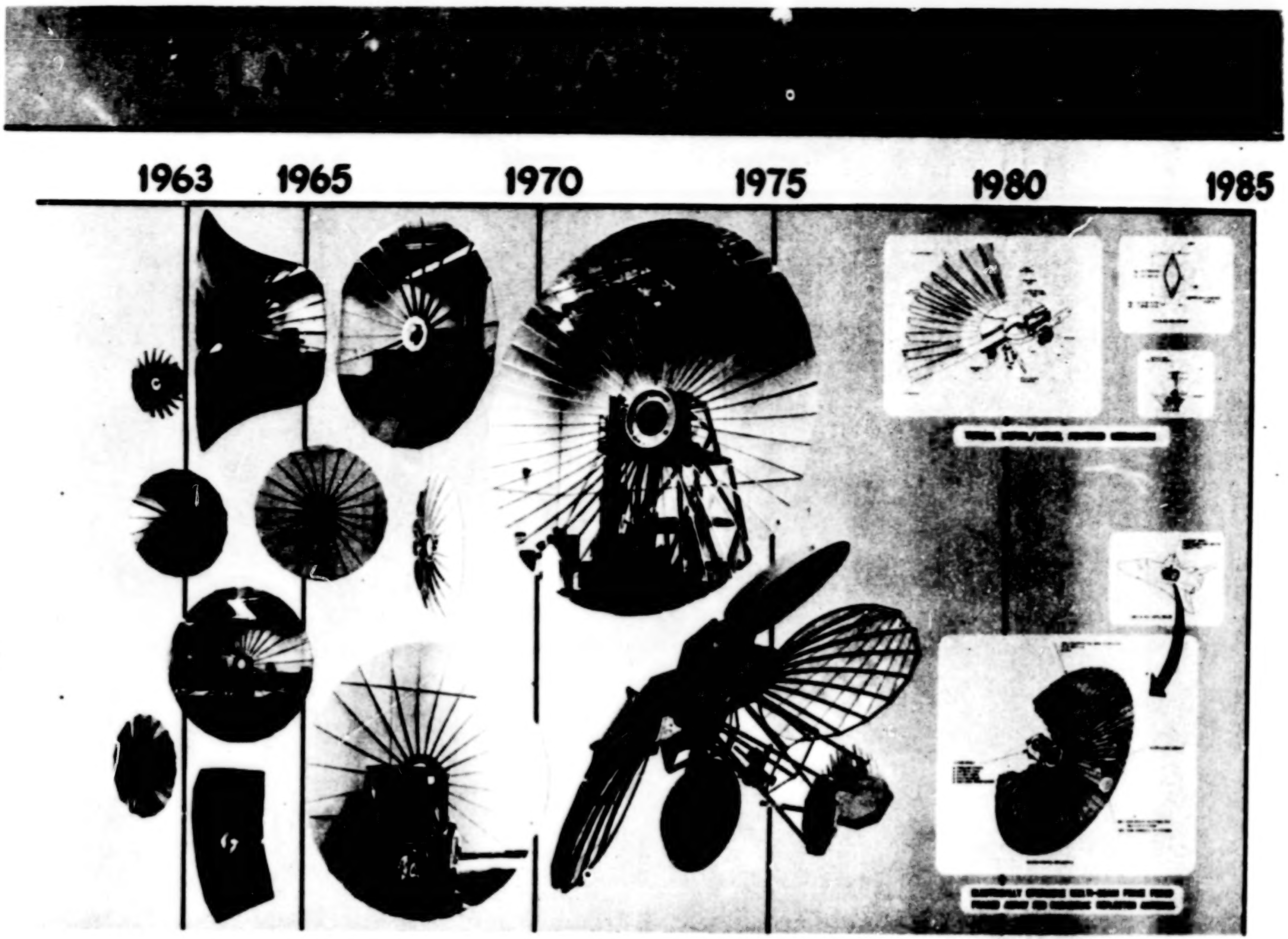


Figure 1

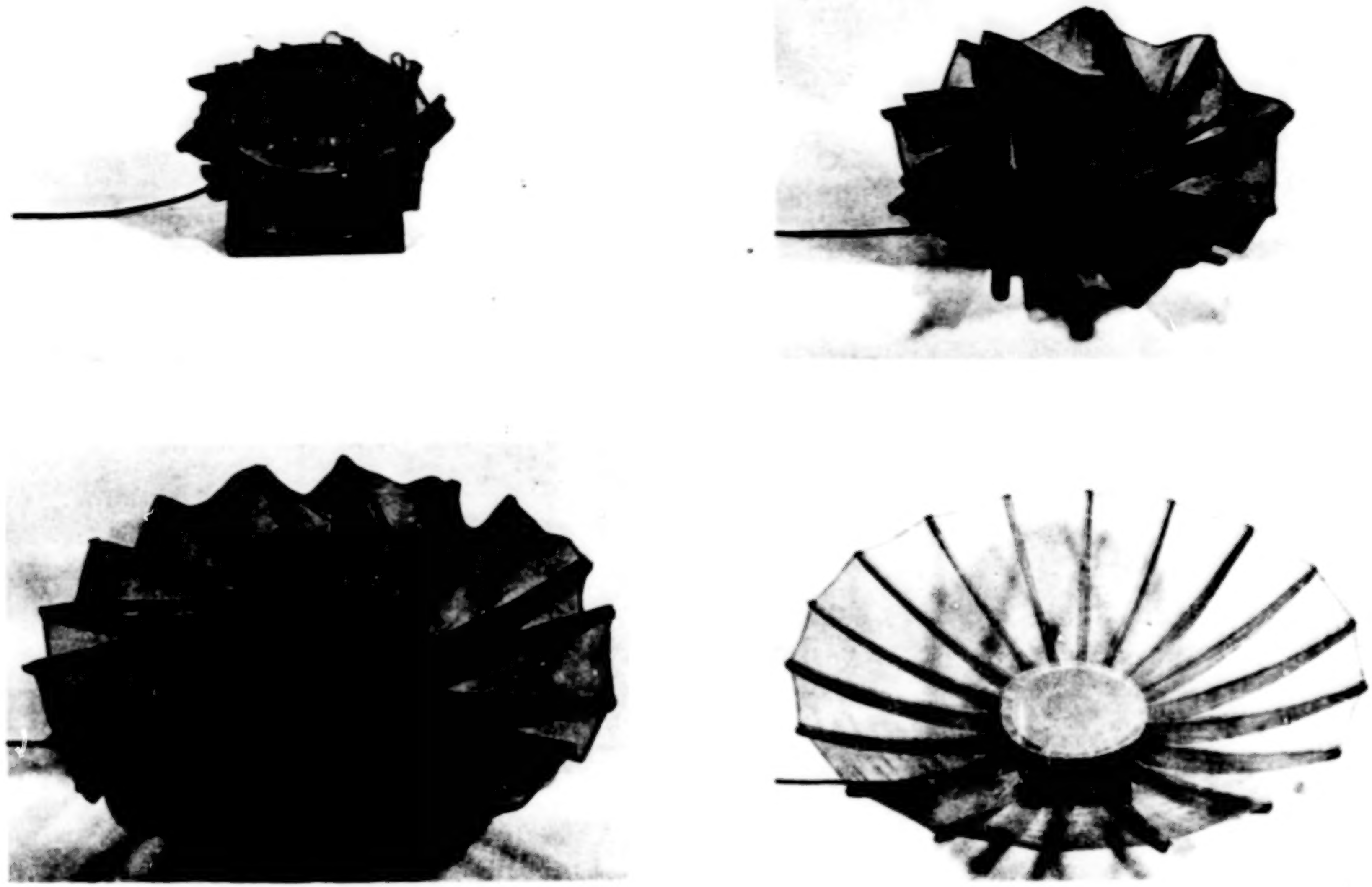
245

246

(Figure 2)

THE WRAP-RIB DESIGN CONCEPT CONSISTS OF A NUMBER (VARIABLE) OF RADIAL RIBS OR BEAMS WHICH ARE CANTILEVERED FROM A CENTRAL HUB STRUCTURE. EACH OF THE RIBS IS ATTACHED TO THIS HUB THROUGH HINGES. THIS RADIAL SPOKE SYSTEM PROVIDES THE MOUNTING FOR THE ANTENNA STRUCTURE. ARRAYS ARE FORMED BY MOUNTING A MEMBRANE WITH ELEMENTS ON THE FRONT EDGE OF THE RIBS AND, IF REQUIRED, A GROUND PLANE ON THE BACK EDGE. FOR PARABOLIC OR OTHER CURVED REFLECTORS, THE RIBS ARE FORMED IN THE REQUIRED SHAPE, AND REFLECTIVE PIE-SHAPED GORES ARE ATTACHED BETWEEN THE RIBS. THE RIB CROSS SECTION AND MATERIAL ARE CHOSEN TO PERMIT THE ELASTIC BUCKLING OF THE RIBS. THIS IS TO ALLOW THE RIBS AND SURFACE TO BE WRAPPED AROUND THE HUB STRUCTURE IN THE ASCENT OR STOWED PACKAGE CONFIGURATION.

THE ELASTIC ENERGY STORED IN THE WRAPPED RIBS IS SUFFICIENT TO ACCOMPLISH DEPLOYMENT OF RELATIVELY SMALL DIAMETER (LESS THAN 20 M) SYSTEMS. IN THIS CASE, THE STOWED PACKAGE IS CONTAINED BY A SERIES OF HINGED DOORS WHICH ARE HELD IN PLACE BY A RESTRAINING CABLE. DEPLOYMENT OCCURS WHEN THE CABLE IS SEVERED. FOR THE LARGER DIAMETERS, THE SURFACE LOADS AND MOMENTUM EXCHANGE WITH THE SPACECRAFT WILL NOT ALLOW THIS FREE DEPLOYMENT. A DEPLOYMENT RESTRAINT SYSTEM HAS BEEN INCORPORATED IN THE DESIGN TO CONTROL THIS SUDDEN RELEASE OF STRAIN ENERGY.



Motor Driven Flex-Rib Antenna Unfurling

Figure 2

247

(Figure 3)

LOCKHEED MISSILES & SPACE COMPANY HAS BEEN INVOLVED IN A PROGRAM TO DEVELOP NEAR-TERM SOLUTION FOR LARGE APERTURE ANTENNAS. THIS PROGRAM WAS BUILT ON THE SUCCESSFUL WRAP-RIB ANTENNA DESIGN CONCEPT. THROUGH THE APPLICATION OF NEW MATERIALS AND DESIGN IMPROVEMENTS, THE WRAP-RIB STRUCTURE CAN NOW BE EXTENDED TO SUPPORT ANTENNAS OF OVER 200 M IN DIAMETER. THESE DESIGNS CAN BE PACKAGED FOR ASCENT ON THE SPACE TRANSPORTATION SYSTEM AND ARE SELF DEPLOYING IN SPACE.

IN ORDER TO ACCOMPLISH THIS DESIGN EVOLUTION, THE WRAP-RIB DESIGN WAS RECEIVED, REQUIRED DESIGN CHARACTERISTICS FOR LARGE DEPLOYABLES IDENTIFIED AND DESIGN SOLUTION IDENTIFIED. WITH THIS COMPLETED, THE BASELINE DESIGN DEVELOPMENT WAS INITIATED ON A 15M ANTENNA. THE ANTENNA IS DESIGNED TO ALLOW CORRELATION BETWEEN GROUND AND ORBITAL TESTING AS A BASIS FOR VERIFYING BASIC SCALING/PERFORMANCE CHARACTERISTICS.

DESIGN TASK UNDERTAKEN

PROBLEM: DEVELOP TECHNOLOGY FOR MANUFACTURING 20 M TO ? SPACE ERECTABLE ANTENNAS

REQUIREMENT	SELECTED APPROACH
ERECT IN SPACE	SELF-ERECTING STORED ENERGY CONTROLLED RELEASE
SCALEABLE 20M TO ?	CANTILEVER BEAM BASIC ELEMENT DEPLOYMENT SYSTEM DESIGN INDEPENDENT OF DIAMETER
HIGH THERMAL STABILITY	COMPOSITE MATERIAL CONSTRUCTION
HIGH DYNAMIC STIFFNESS	COMPOSITE (STIFFNESS/WEIGHT) LENTICULAR CROSS SECTION RIB
MINIMUM STOWED SIZE	WRAP-RIB DESIGN

Figure 3

250

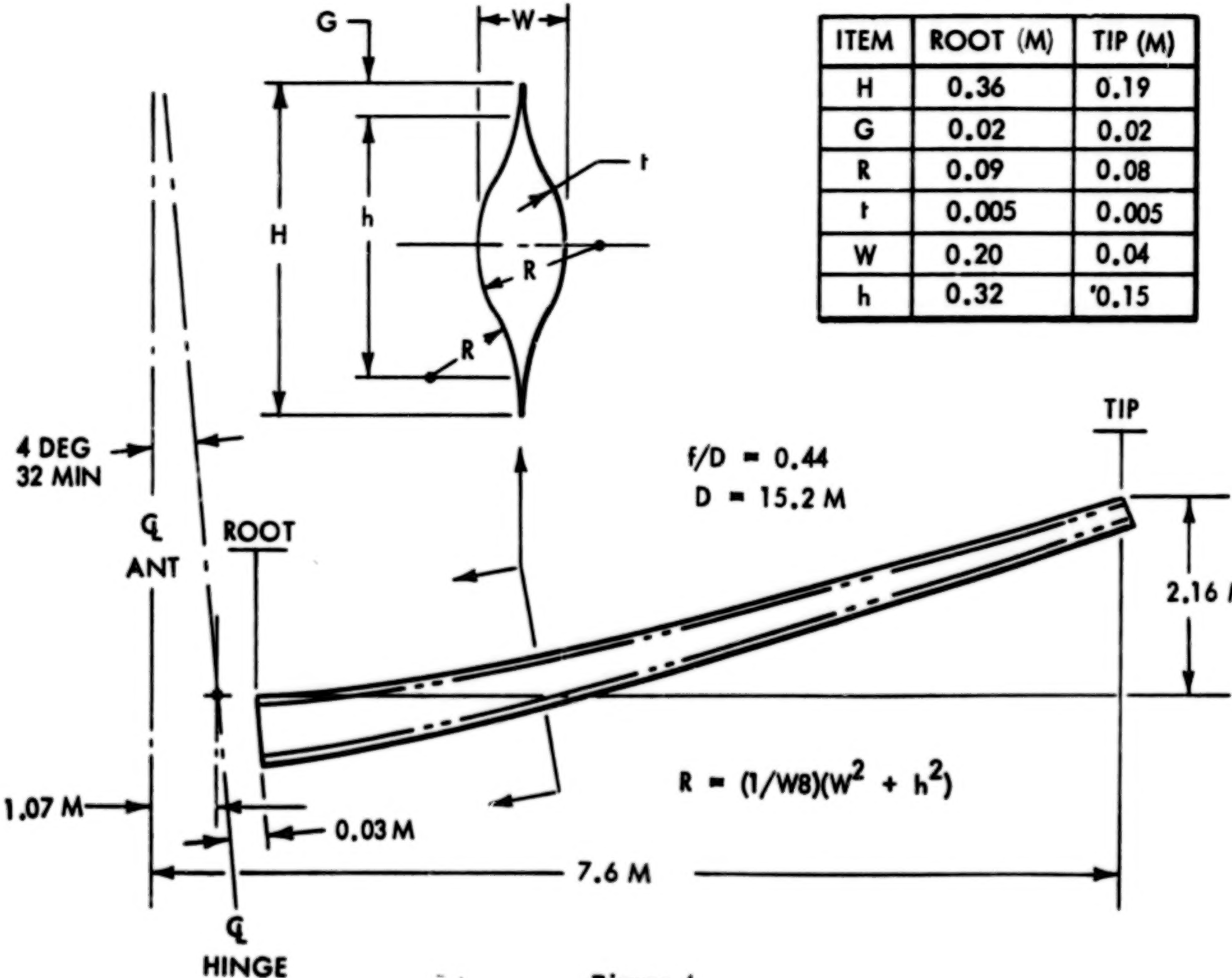
(Figure 4)

PREVIOUSLY, THE RIBS HAVE BEEN FABRICATED FROM ALUMINUM. THE CROSS-SECTION WAS SEMILENTICULAR (C-SHAPED) WITH CHEMICALLY ETCHED RADIAL STIFFENERS. THIS DESIGN WAS LIMITED IN SIZE BY THE HIGH THERMAL COEFFICIENTS OF EXPANSION AND THE DENSITY OF THE METAL. THE DENSITY LIMITED THE AVAILABLE DIAMETER BY WEIGHT, AND THE THERMAL COEFFICIENT OF EXPANSION LIMITED THE ANTENNA FREQUENCY SINCE EXCESSIVE ORBIT SURFACE DISTORTIONS OCCURRED EVEN WITH OPTIMIZED THERMAL CONTROL SYSTEMS.

TO REMOVE THESE CONSTRAINTS, LMSC INVESTIGATED THE USE OF COMPOSITES FOR THE RIB APPLICATION. AFTER SEVERAL YEARS OF DEVELOPMENT, IT HAS BEEN DETERMINED THAT GRAPHITE EPOXY RIBS CAN BE FABRICATED AND CAN PERFORM TO THE REQUIREMENTS. RIBS HAVE NOW BEEN FABRICATED TO REPLACE THE SEMI-LENTICULAR DESIGN AND ARE EVEN BEING FABRICATED IN A FULL LENTICULAR CROSS SECTION.

THE DESIGN DETAILS FOR A 7.2-M-LONG GRAPHITE EPOXY LENTICULAR RIB ARE PRESENTED IN THE FIGURE.

RIB CONFIGURATION



ITEM	ROOT (M)	TIP (M)
H	0.36	0.19
G	0.02	0.02
R	0.09	0.08
t	0.005	0.005
W	0.20	0.04
h	0.32	'0.15

Figure 4

252

(Figure 5)

CENTRAL HUB DESIGN. SEVERAL OBJECTIVES WERE ESTABLISHED FOR THE CONTROLLED DEPLOYMENT OF LARGE REFLECTORS.

1. THE SYSTEM SHOULD BE MECHANICALLY SIMPLE

2. THE SYSTEM SHOULD RESTRAIN THE RELEASE OF ENERGY AS OPPOSED TO ACTIVELY DEPLOYING THE REFLECTOR

3. THE SYSTEM SHOULD BE COMPATIBLE WITH BOTH CURVED AND STRAIGHT RIBS

4. THE SYSTEM SHOULD ALLOW THE ATTACHMENT OF MORE THAN ONE MEMBRANE TO THE RIBS

5. THE SYSTEM SHOULD BE SIZE INDEPENDENT

A DESIGN SOLUTION WHICH SATISFIES THESE OBJECTIVES FOR A DEPLOYMENT RESTRAINT SYSTEM WAS FOUND. THE SOLUTION USES A TAPE AND PULLEY SYSTEM. WITH THIS SYSTEM, A TAPE IS PLACED BETWEEN EACH RIB, THE RIBS AND MESH ARE WRAPPED WITH THE TAPE, AND THE TAPE, UNDER TENSION, KEEPS EACH RIB WRAPPED AROUND THE HUB. FOR DEPLOYMENT A MOTOR DRIVES A LARGE GEAR WHICH IN TURN ROTATES THE TAPE TAKE-UP REELS. THE RIBS DEPLOY AS THE TAPE IS REELED UP. A CONSTANTLY SLIPPING CLUTCH IS LOCATED IN EACH TAPE REEL DRIVE TO LIMIT THE TAPE TENSION AND ALLOW POSITIONAL AND SPEED VARIATIONS FROM RIB TO RIB AND DURING REFLECTOR DEPLOYMENT. THE POWERED TAPE OPERATED DEPLOYING MECHANISM HAS BEEN FABRICATED AND INCLUDED IN THE LARGE APERTURE DEMONSTRATION MODEL. BECAUSE THE FUNCTION OF THE TAPE IS TO APPLY PRESSURE TO THE RIB, THE SYSTEM ALLOWS MEMBRANES TO BE WRAPPED BETWEEN THE TAPE AND THE RIB AND THUS PERMITS LOCATING SURFACE ATTACHMENTS ANYWHERE ON THE RIB. THE DESIGN IS ALSO INDEPENDENT OF DIAMETER BECAUSE LARGER DIAMETERS WILL ONLY INCREASE TAPE PULLEY DIAMETER AND OPERATING TIME.

POWERED TAPE OPERATED DEPLOYING MECHANISM

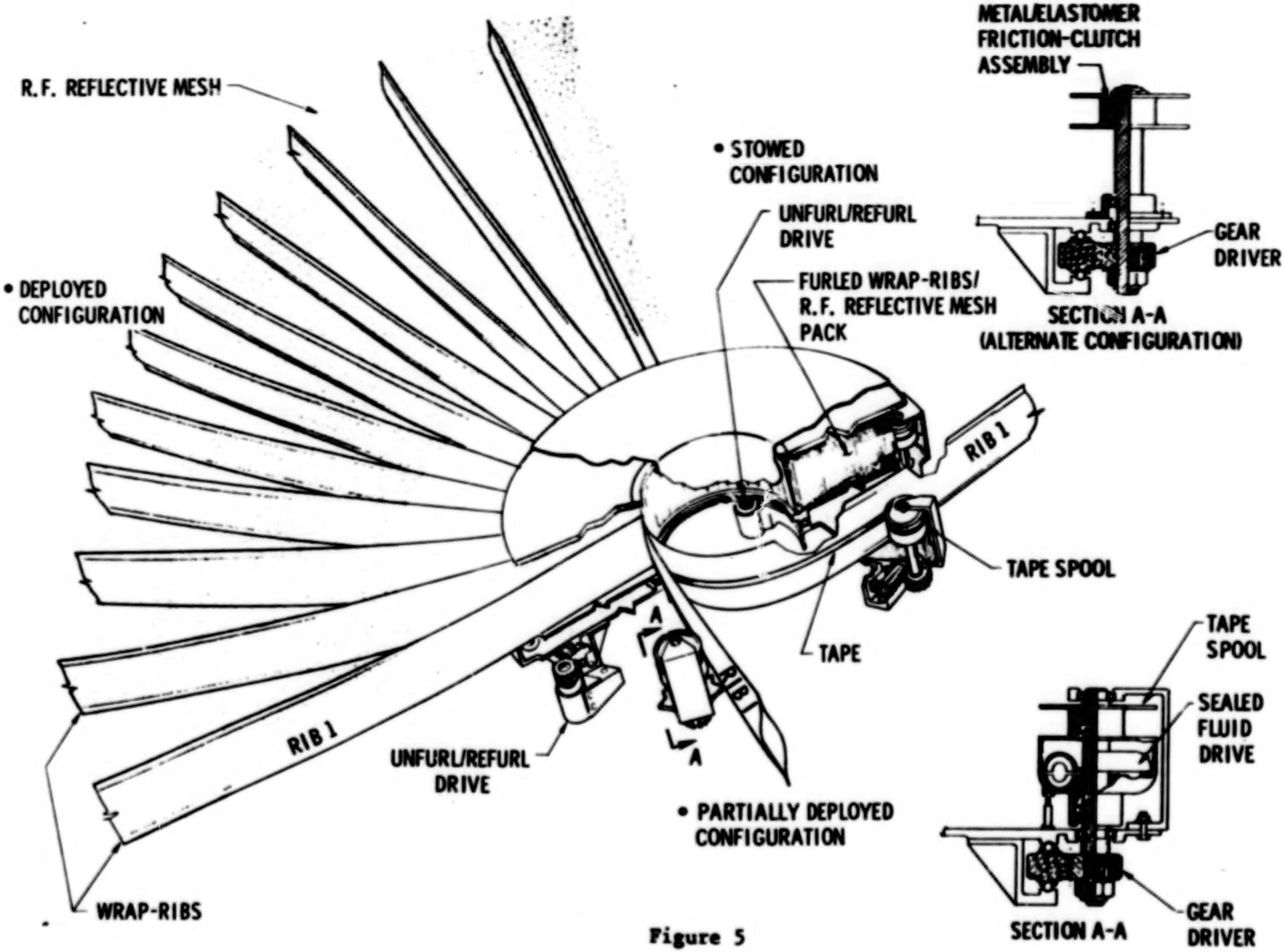


Figure 5

253

254

(Figure 6)

THE RIB HAS BEEN FABRICATED AND IS SHOWN NEXT TO THE RIB TOOLING IN THE FIGURE. THE COMPLETED RIB WEIGHT WAS 3.3 KG. FUNDAMENTAL NATURAL FREQUENCIES WERE 5 AND 10 Hz ABOUT THE WEAK AND STRONG AXES, RESPECTIVELY.

LENTICULAR RIBS HAVE EXHIBITED VISCOELASTIC BEHAVIOR; HOWEVER, RESULTING CROSS-SECTIONAL DEFORMATIONS DURING STORAGE WERE SMALL AND DID NOT AFFECT THE DEPLOYMENT CHARACTERISTICS OF THE RIB. IN ALL CASES, APPROXIMATELY 80 PERCENT OF THE VISCOELASTIC DEFORMATIONS WERE RECOVERABLE DURING A PERIOD OF SEVEN DAYS. THIS HIGH DEGREE OF RECOVERY FROM VISCOELASTIC EFFECTS AND THE ABILITY TO MINIMIZE THESE EFFECTS BY PROPER LAY-UP DESIGN AND ENVIRONMENTAL CONTROL ENSURE DIMENSIONAL STABILITY AFTER DEPLOYMENT.

WRAP RIB AND TOOLING

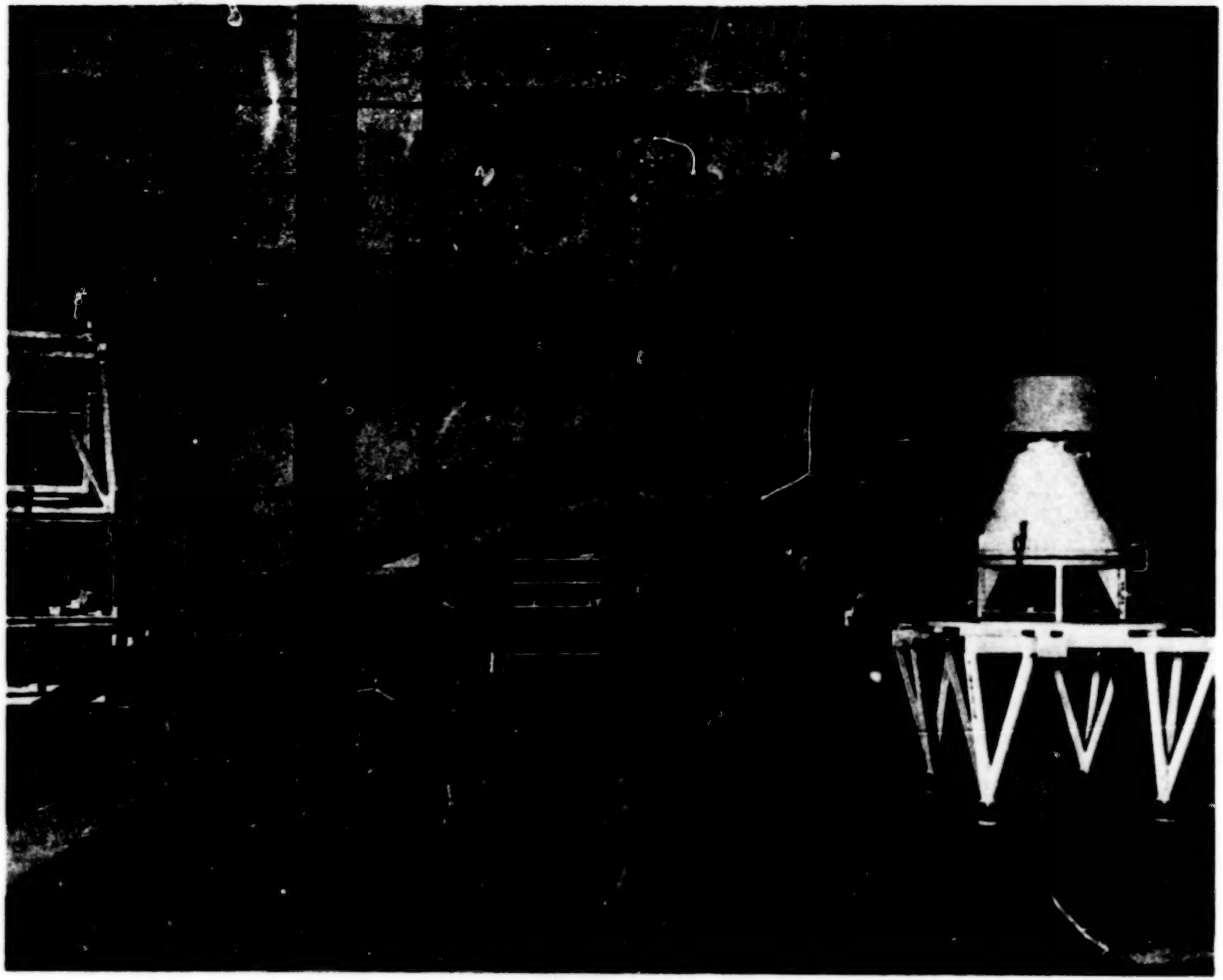
Figure 6

255

256

(Figure 7)

THE TEST RIB HAS BEEN INSTALLED ON A HUB WITH A MOTOR DRIVE SYSTEM PROTOTYPE. THE RIB HAS BEEN STOWED AND DEPLOYED MORE THAN 60 TIMES WITH NO EVIDENCE OF DEGRADATION. DEPLOYMENT OR RETRACTION OF THE RIB IS ACCOMPLISHED IN LESS THAN 2 MINUTES.

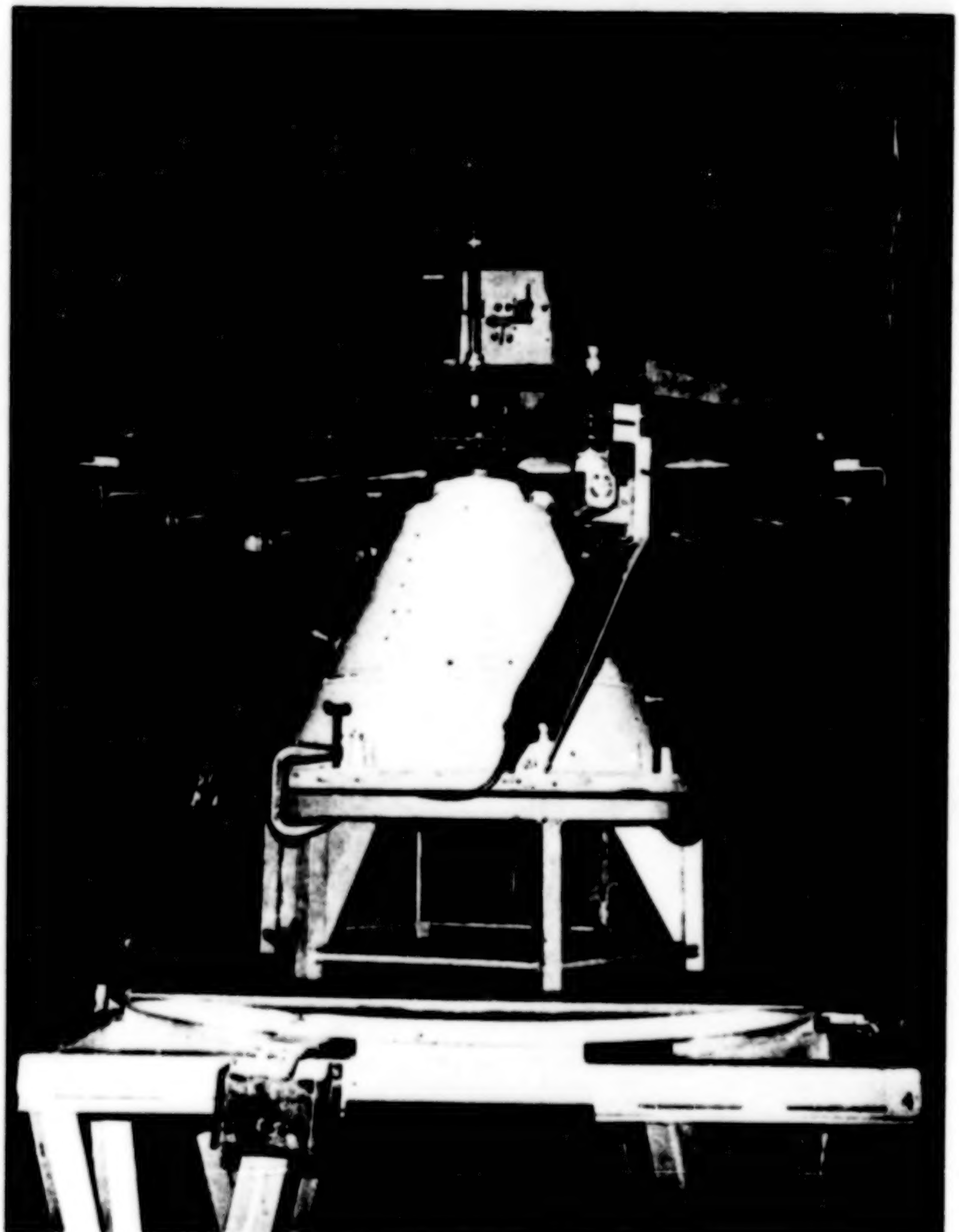


RIB INSTALLED ON DEPLOYMENT HUB
Figure 7

258

(Figure 8)

THE FULLY STOWED RIB IS HELD IN PLACE BY THE DEPLOYMENT RESTRAINT SYSTEM TAPE. AS THIS TAPE IS ROLLED UP ON THE PULLEY, THE RIB STORED ENERGY FORCES THE RIB TO EXTEND AND FORM.



FULLY STOWED RIB ON DEPLOYMENT HUB

Figure 8

(Figure 9)

A PRELIMINARY SYNCHRONOUS ORBIT THERMAL DISTORTION ANALYSIS HAS BEEN PERFORMED. THE THEORETICAL PROPERTIES OF THE DESIGN LAYUP WERE USED IN THE ANALYSIS AND A MULTI-LAYER INSULATION BLANKET WAS ASSUMED TO BE COVERING THE RIB. THE ANALYSES RESULTS WERE THEN USED TO ESTABLISH THE DIAMETER LIMIT AS A FUNCTION OF MAXIMUM ON AXIS GAIN LOSS ALLOWED AND THE ANTENNA OPERATIONAL FREQUENCY.

ORBIT THERMAL DISTORTION APERTURE LIMITATIONS

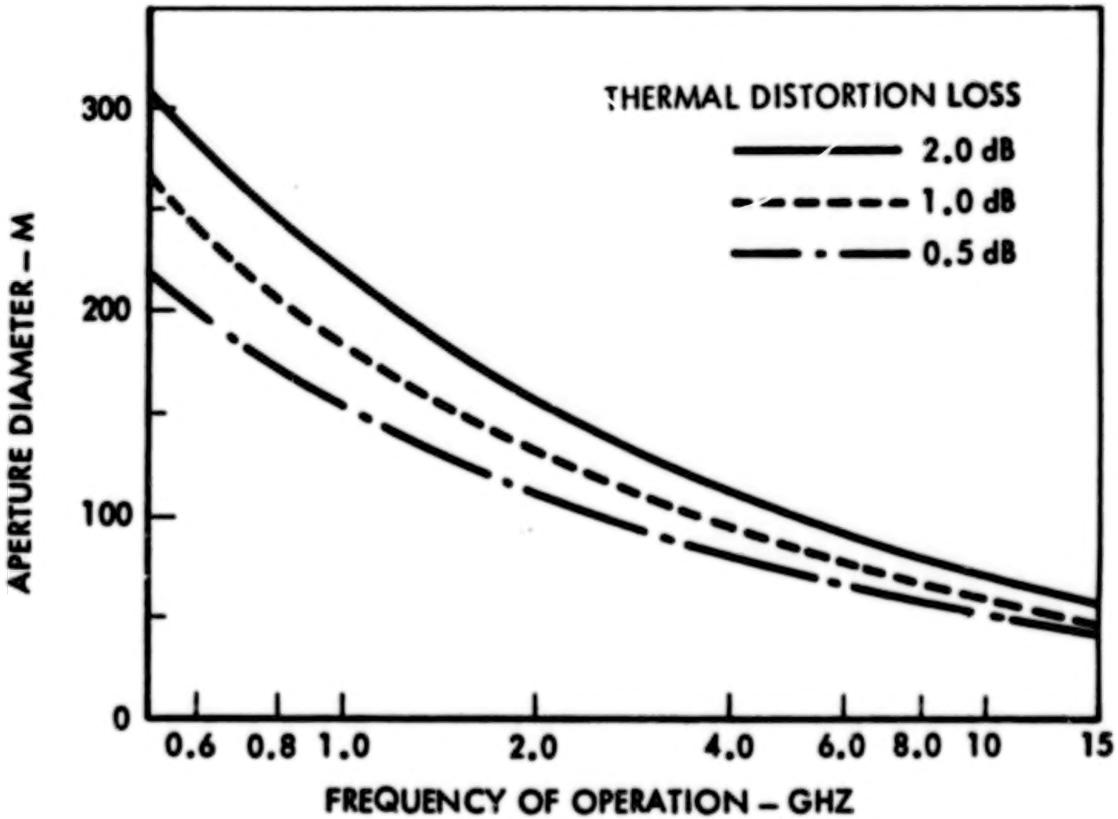


Figure 9

261

(Figure 10)

WITH PRELIMINARY DIAMETER AND FREQUENCY LIMITS ESTABLISHED WEIGHT AND STOWED PACKAGE SIZE CAN BE DETERMINED. THESE CHARACTERISTICS HAVE BEEN IDENTIFIED FOR A SYSTEM WITH A FOCAL LENGTH TO DIAMETER (F/D) RATIO OF 1.5 AND A PARABOLIC APPROXIMATION ERROR OF 0.5 dB. IN THE FORMULATION OF THE DATA THE MAXIMUM OPERATION FREQUENCY WAS USED AT ALL DIAMETERS TO PROVIDE AN UPPER BOUND.

WEIGHT & STOWED DIAMETER CHARACTERISTICS

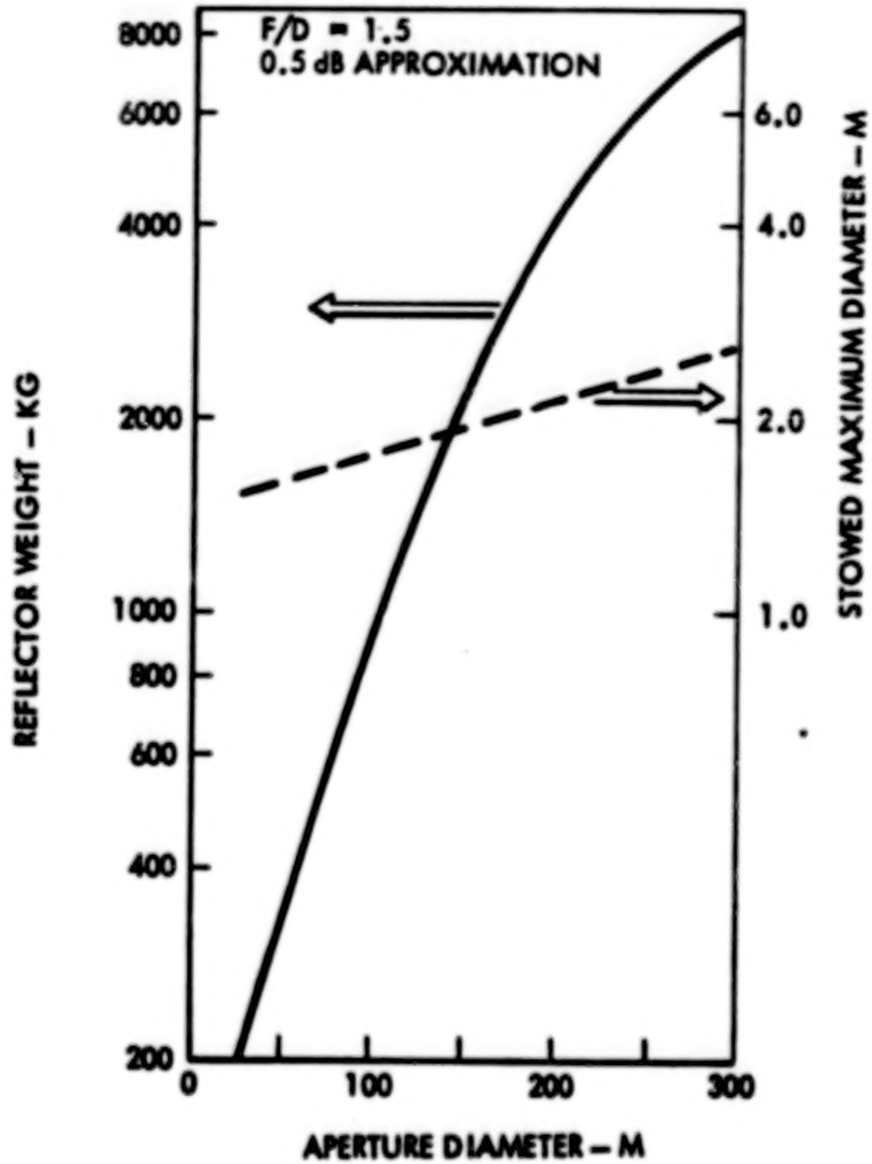


Figure 10

263

264

(Figure 11)

THE DEPLOYED NATURAL FREQUENCIES OF THE REFLECTORS WERE ALSO CALCULATED. THESE FREQUENCIES ARE CONSISTENT WITH THE WEIGHT AND STOWED DIAMETERS PRESENTED. IF IT IS DESIRED TO INCREASE NATURAL FREQUENCY ABOVE THESE PREDICTIONS, IT CAN BE ACCOMPLISHED BY CHANGING THE RIB CROSS-SECTIONS GEOMETRY. THIS WOULD RESULT IN A WEIGHT INCREASE ONLY. THE STOWED PACKAGE SIZE WOULD REMAIN THE SAME.

STRUCTURAL DYNAMICS CHARACTERISTICS

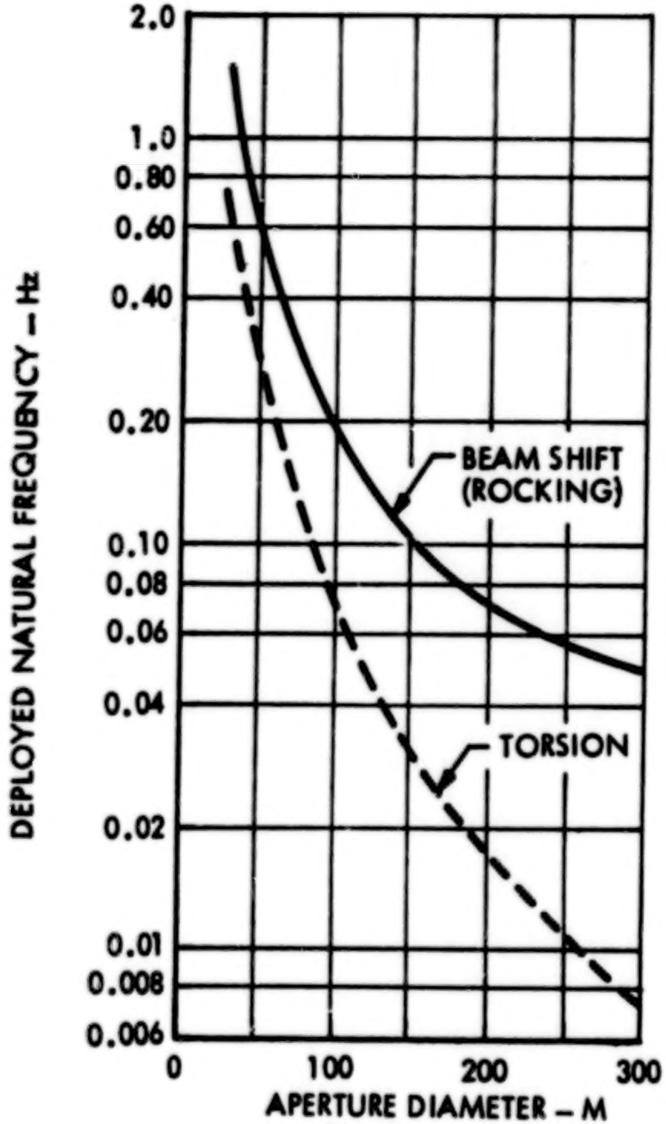


Figure 11

265

(Figure 12)

THE CORRELARY OF HIGH GAIN, LARGE WAVELENGTH APERTURES IS A NARROW BEAM SMALL EARTH SPOT COVERAGE. LARGER FIELDS OF VIEW MAY BE PROVIDED BY MULTIPLE BEAM FEED SYSTEMS CONSISTING OF N ELEMENTS. SUCH SYSTEMS CAN AUTOMATICALLY CORRECT FOR OFF-AXIS DEGRADATION. A TRANSFORMING MATRIX IS USED TO TRANSFORM THE SMALL FEED ELEMENT APERTURE TO AN APERTURE SIZE APPROXIMATELY EQUAL TO 1/N OF THE DISH APERTURE. THE RESULTANT "NEW ELEMENT" MAY NOW BE PROCESSED IN THE SAME MANNER THAT A PHASED ARRAY IS PROCESSED. THIS TECHNIQUE REPRESENTS A NEAR IDEAL COMPROMISE BETWEEN THE SIMPLICITY OF LARGE PARABOLIC STRUCTURES AND THE VIRTUES OFFERED BY PHASED ARRAYS.

MULTIPLE-BEAM FEED BLOCK DIAGRAM

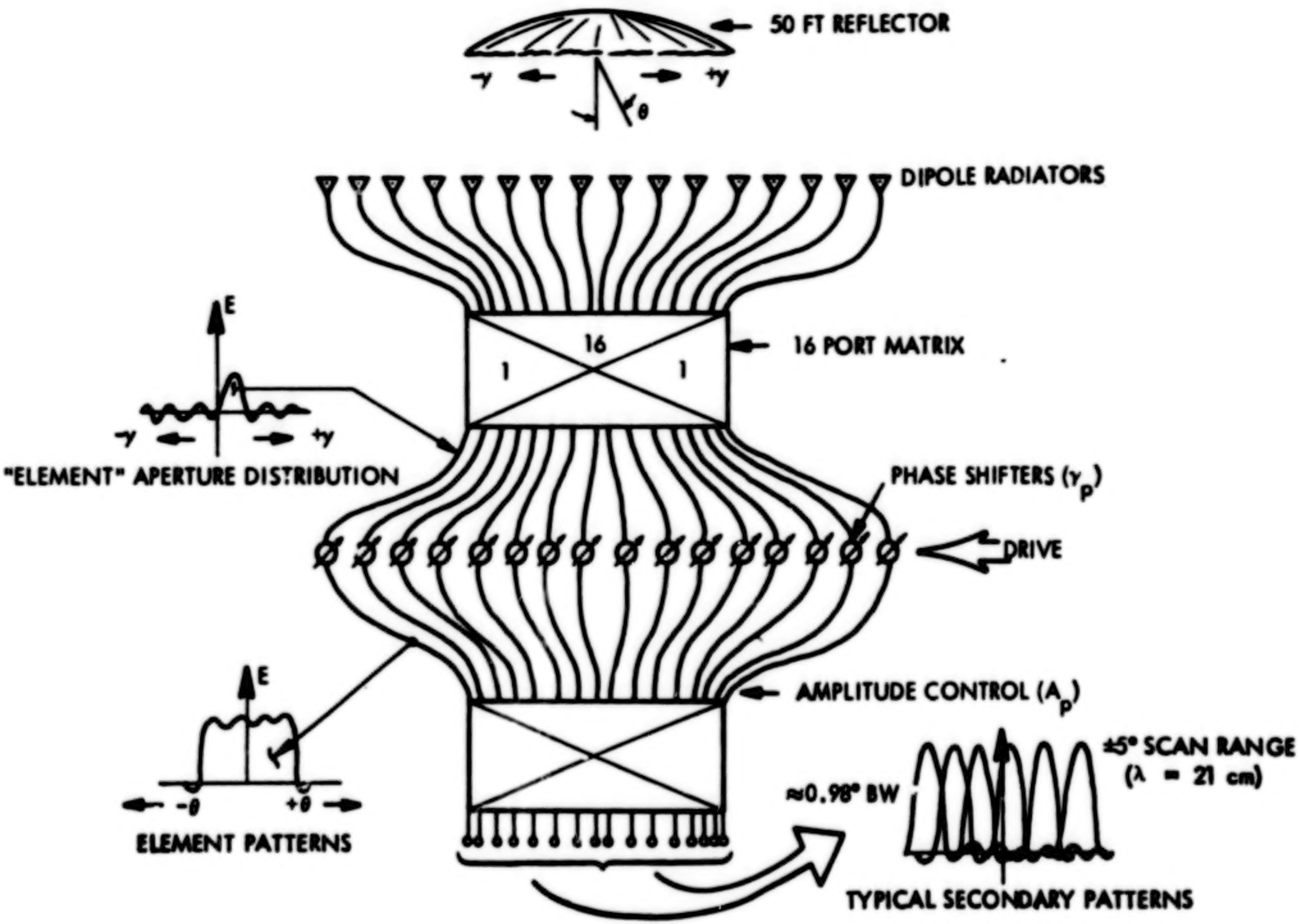


Figure 12

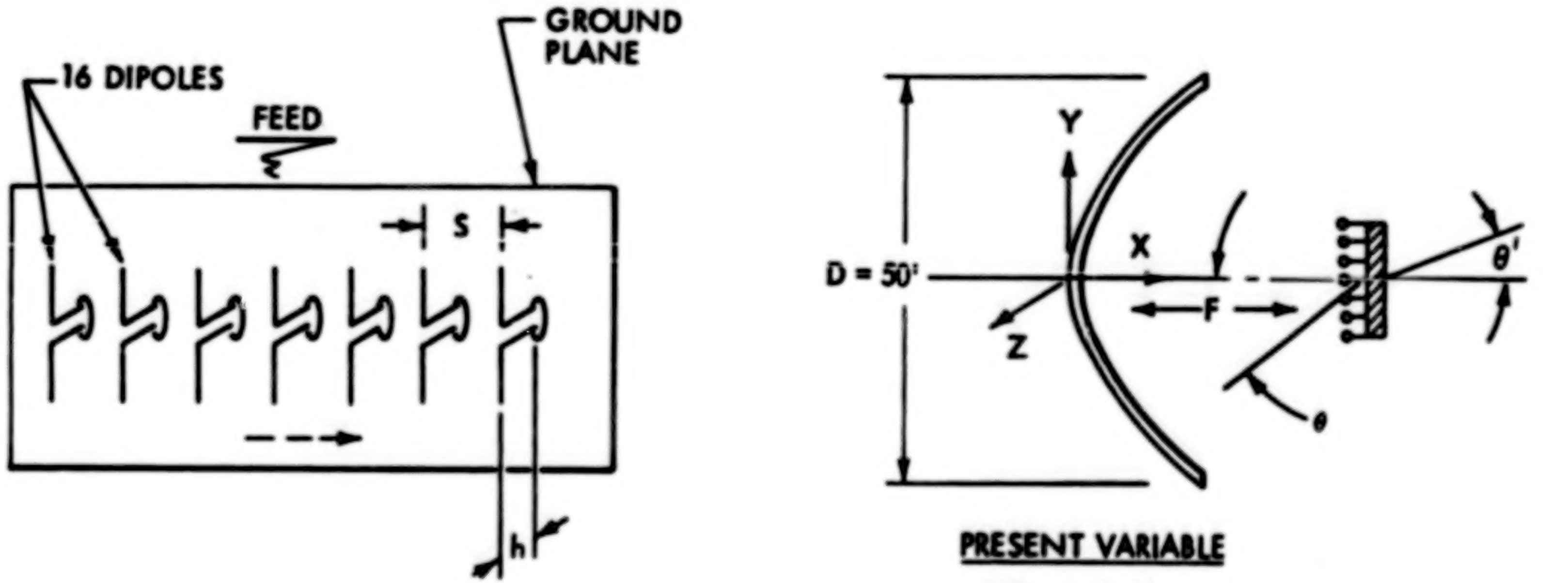
267

268

(Figure 13)

A PRACTICAL WAY TO ILLUSTRATE THE THEORY IS TO USE A SIMPLE EXAMPLE INVOLVING A 16 ELEMENT LINE SOURCE FEED EXCITING A 50 FT. PARABOLIC DISH AT A WAVELENGTH OF 1 CM.

TRIAL CASE



PRESENT VARIABLE

$F/D = 0.44$
$S/\lambda = 0.582$
$h/\lambda = 0.200$
A_p (p) = ELEMENT NO.
γ_p (PHASE SHIFT FOR P^{TH} ELEMENT
$\lambda = 21$ CM

Figure 13

269

(Figure 14)

THE 16 BEAMS PRODUCED BY THE TRANSFORMED MATRIX, PROVIDE 16 SEPARATELY EXCITED ZONES ON THE PARABOLIC DISH. THESE EXCITATION ZONES MAY BE THEORETICALLY COMPRESSED INTO AN EQUIVALENT LINE SOURCE, EACH SHOWING THE DISTRIBUTION IN THE APERTURE OF THE DISH IN THE DIAMETER OF THE SCAN PLANE FOR THE PARTICULAR BEAM IN QUESTION. THE SLIDE SHOWS THE DISTRIBUTIONS FOR THE FIRST AND FIFTH MOST DISPLACED BEAMS FROM THE DISH APERTURE.

EQUIVALENT LINE SOURCE EXCITATIONS FOR ELEMENTS 1 & 5

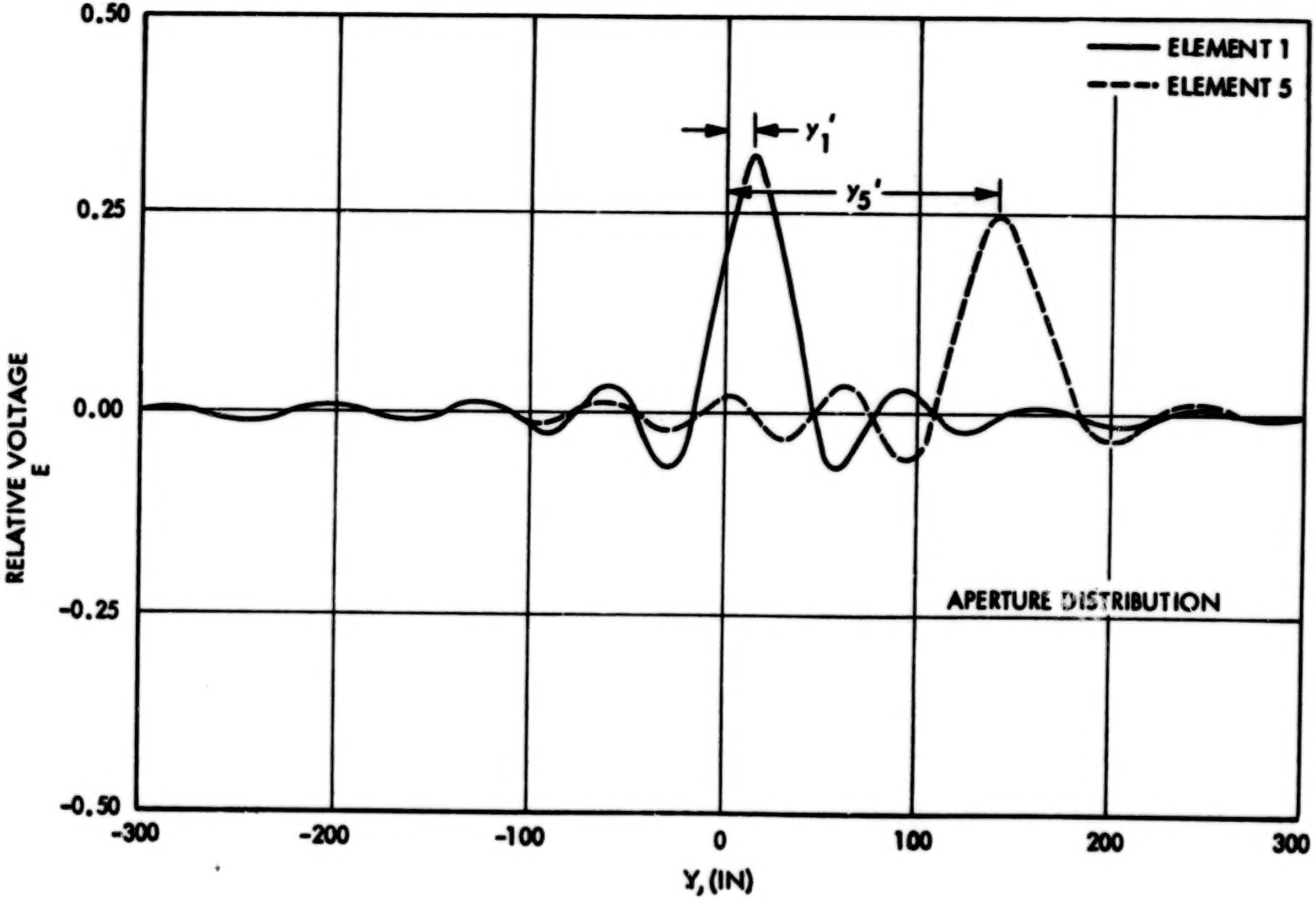


Figure 14

272

(Figure 15)

THIS SLIDE SHOWS THE COMPOSITE OF ALL APERTURE DISTRIBUTIONS IN THE EQUIVALENT LINE SOURCE ALONG THE DIAMETER WHICH LIES IN THE SCAN PLANE. NOTE THAT THE APERTURE DISTRIBUTIONS OVERLAP. IT IS THIS OVERLAPPING WHICH ALLOWS THE HIGH CROSSOVER LEVELS FOR THE SECONDARY MULTIPLE BEAMS.

EQUIVALENT LINE SOURCE SUMMATION OF ALL DISTRIBUTIONS

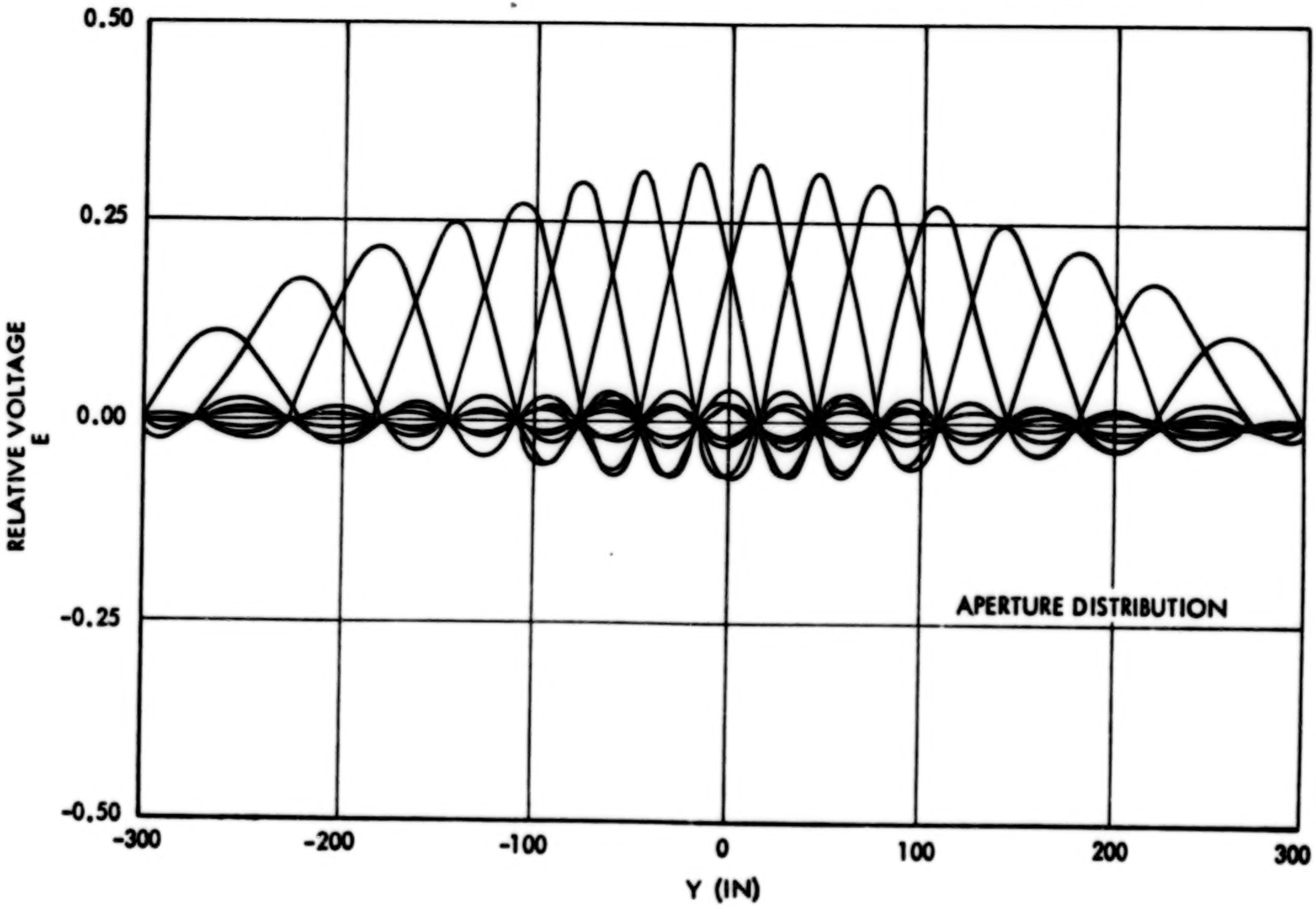


Figure 15

(Figure 16)

EACH APERTURE DISTRIBUTION PRODUCES ITS OWN ELEMENT PATTERN. THE SLIDE SHOWS THE RESULTANT "NEW ELEMENT PATTERN" FOR THE SECOND MOST REMOVED FEED BEAM FROM THE FOCAL AXIS. THE NEAR SQUARE (AND HENCE NEAR IDEAL) ELEMENT PATTERN RESULTS FROM THE SIDE LOBE LEVELS IN THE DISTRIBUTIONS SHOWN EARLIER.

EQUIVALENT ELEMENT PATTERN

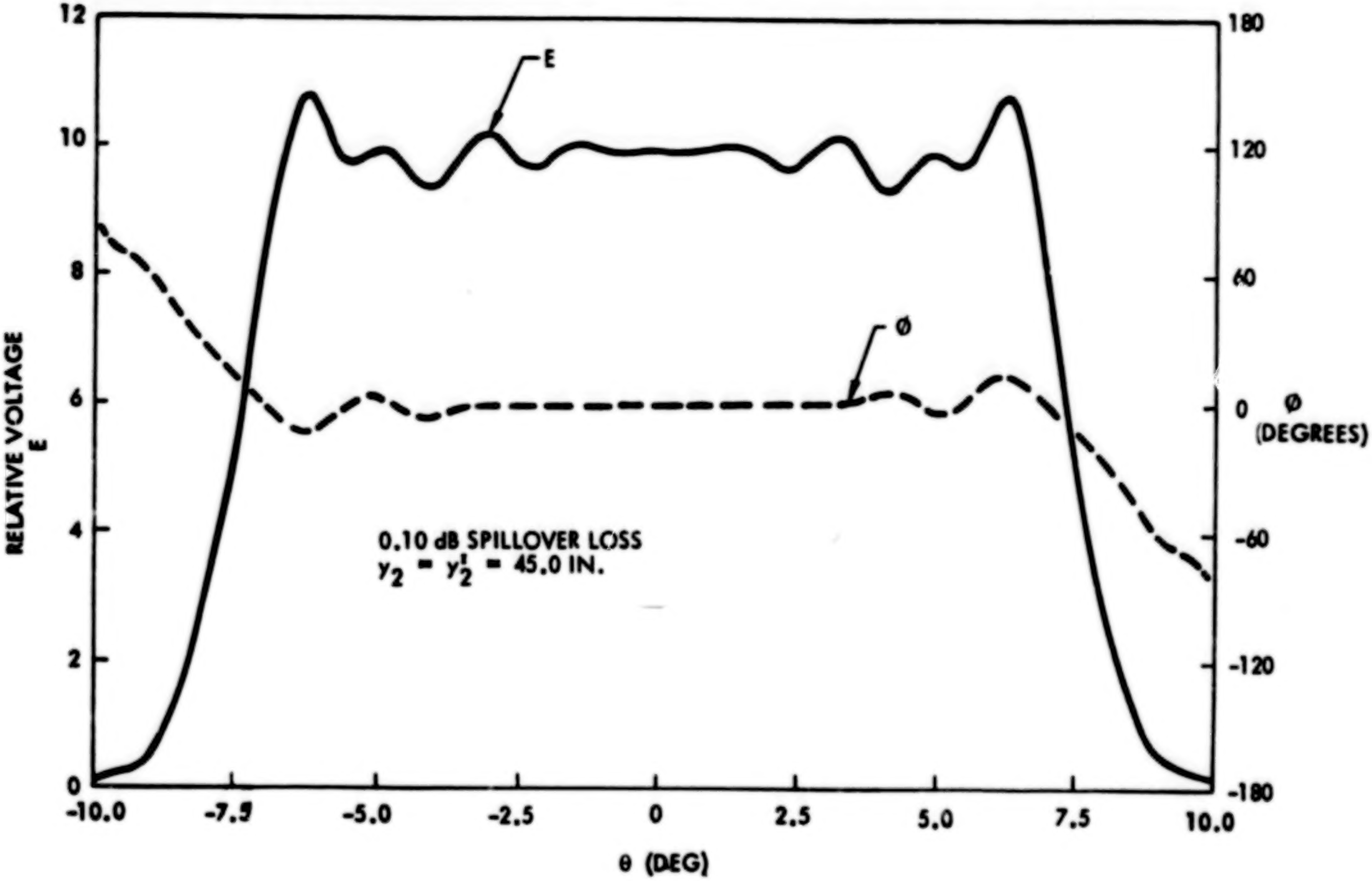


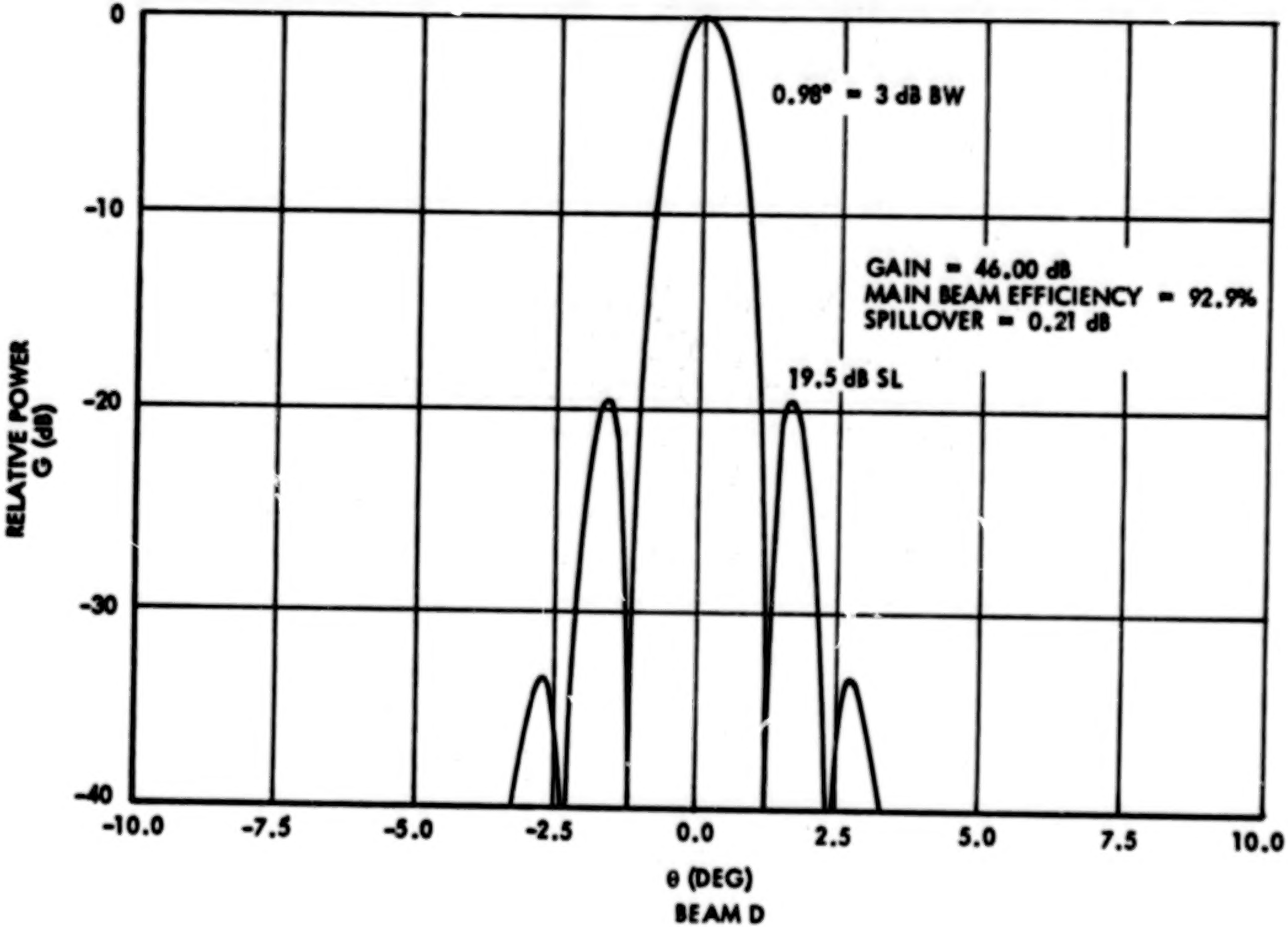
Figure 16

276

(Figure 17)

THE RESULTANT ONE-AXIS SECONDARY BEAM FOR UNIFORM ILLUMINATION (A_p = 1) IS SHOWN IN THIS SLIDE. SOME ILLUMINATION TAPERING IS ACHIEVED BY VIRTUE OF THE H-PLANE ELEMENT PATTERN TAPER AS WELL AS THE SPACE ATTENUATION. THE 19.5 dB SIDE LOBE LEVEL COULD BE IMPROVED BY TAPERING THE DISTRIBUTION COEFFICIENTS (A_p).

SECONDARY PATTERN (ON AXIS)



277

Figure 17

CONTENTS

278

(Figure 18)

THIS SLIDE SHOWS THE SECONDARY BEAM WITH FOUR BEAMWIDTHS DISPLACEMENT FROM THE FOCAL AXIS. THERE IS NO EVIDENCE OF A COMA LOBE AND THE SIDE LOBE LEVEL HAS ACTUALLY IMPROVED BY APPROXIMATELY 1 dB. THE GAIN IS WITHIN 0.1 dB OF THE ON-AXIS GAIN.

SECONDARY PATTERN (4.0BW OFF AXIS)

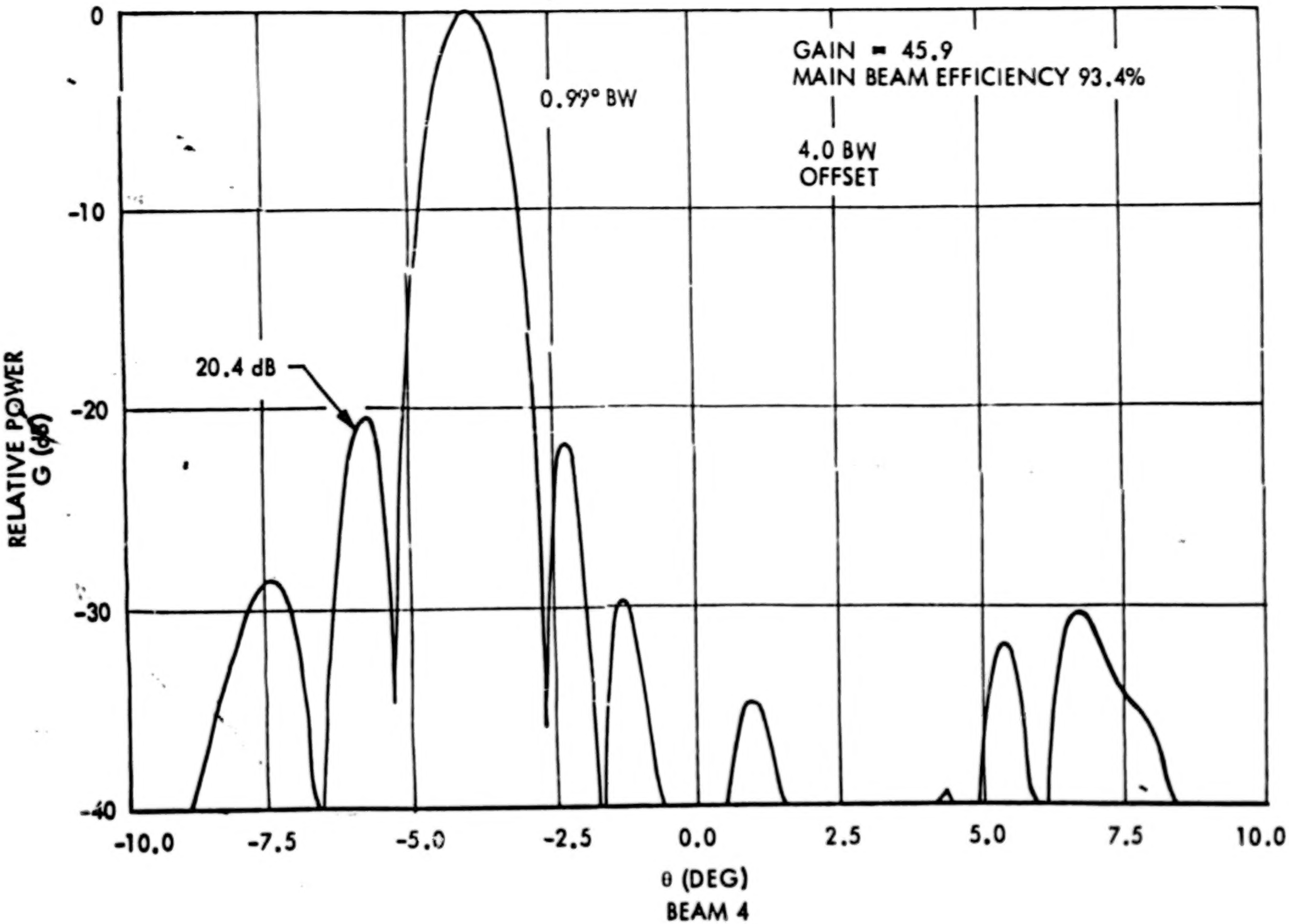


Figure 18

279

280

(Figure 19)

THE COMPOSITE COVERAGE PROVIDED BY 11 SIMULTANEOUS BEAMS IS SHOWN IN THIS SLIDE. ADDITIONAL BEAMS EXIST BUT ARE OF LOWER QUALITY.

SIMULTANEOUS COVERAGE FOR 11 CENTRAL MULTIPLE BEAMS

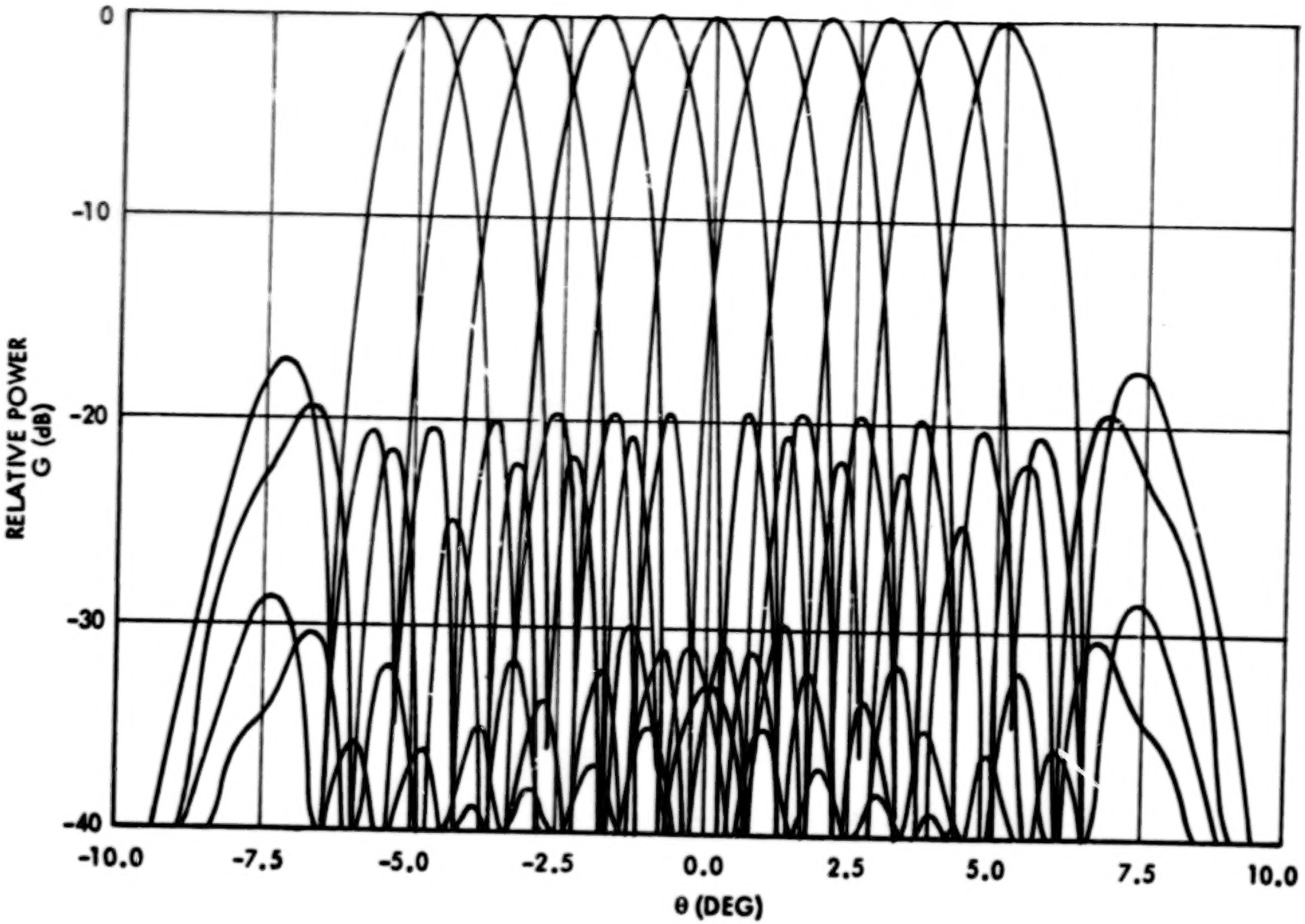


Figure 19

(Figure 20)

THE ELEVATION OR NON-SCAN PLANE PATTERNS ARE OF HIGH QUALITY AND ESSENTIALLY INVARIANT WITH SCAN POSITION. THIS SLIDE SHOWS THE ELEVATION PLANE PATTERN FOR A SCAN WHICH IS 4 BEAMWIDTHS FROM THE FOCAL AXIS.

ELEVATION PLANE PATTERN (4.0 BW SCAN IN AZIMUTH)

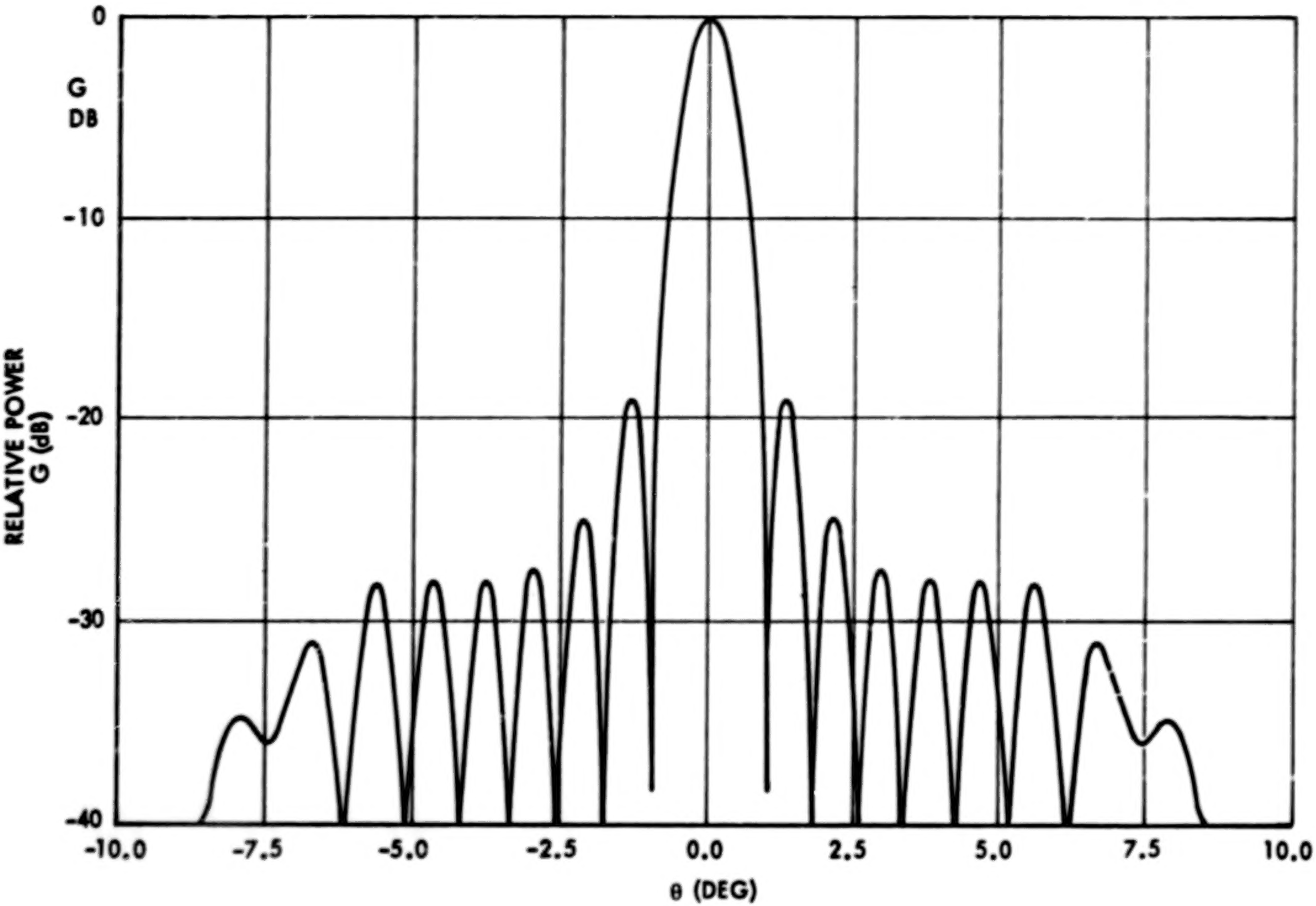


Figure 20

284

(Figure 21)

THE GAIN AND MAIN BEAM EFFICIENCY ARE SHOWN IN THIS SLIDE. THE GAIN IS WITHIN 0.2 dB OF THE ON-AXIS GAIN OVER 8 BEAMWIDTHS COVERAGE. THE BEAM EFFICIENCY IS ALMOST CONSTANT OVER 5 BEAMWIDTHS OF COVERAGE ON EACH SIDE OF THE MAIN AXIS.

GAIN & BEAM EFFICIENCY VS SCAN ANGLE

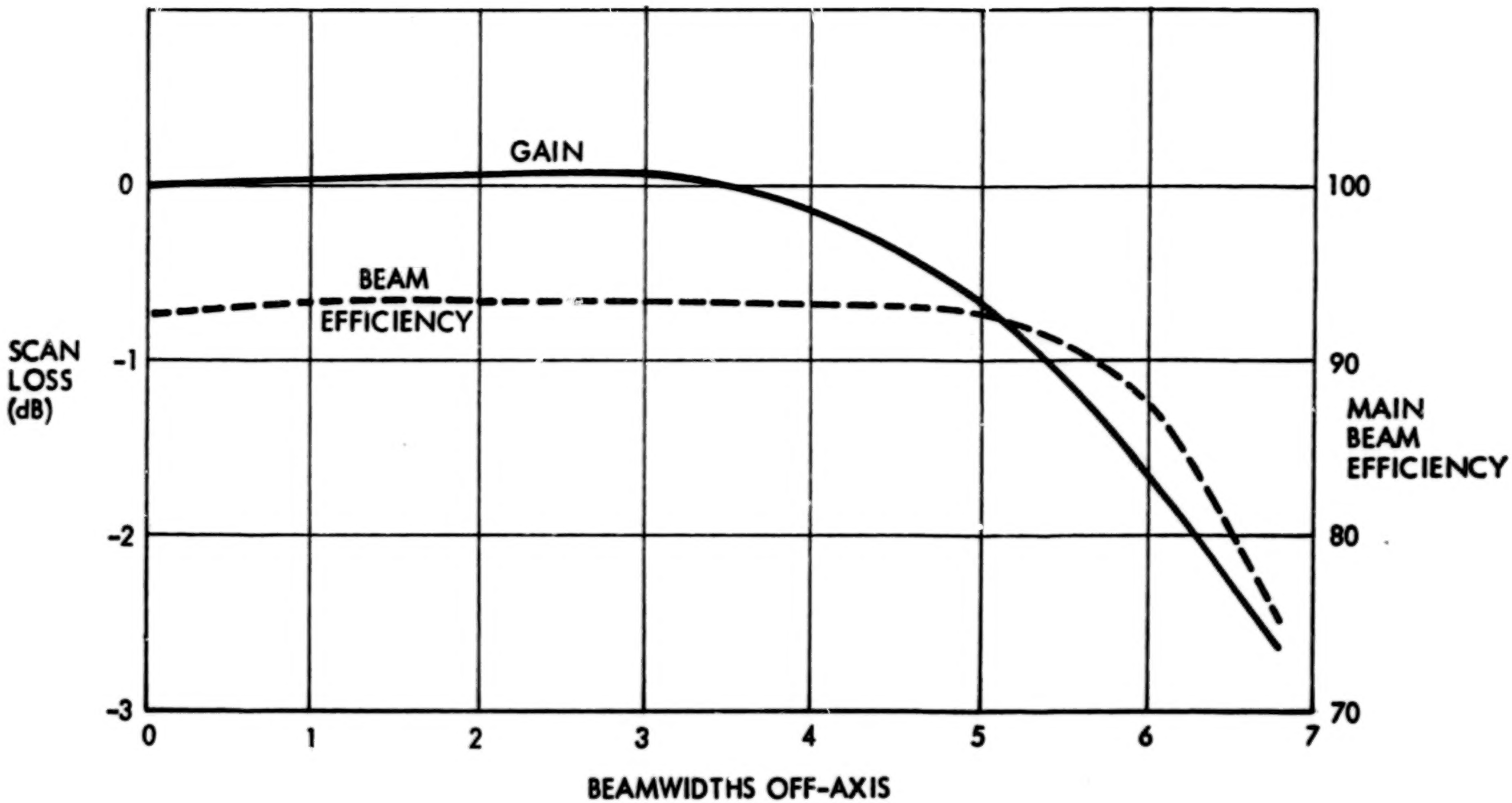


Figure 21

(Figure 22)

A TOTAL OF 11 BEAMS FROM A 16 ELEMENT FEED ARRAY PROVIDE HIGH QUALITY PERFORMANCE. THE NUMBER OF AVAILABLE BEAMS OF GOOD PERFORMANCE WILL REDUCE AS THE FEED SYSTEM IS DISPLACED FROM THE AXIS. FOR EXAMPLE, IF THE FEED SYSTEM WERE CENTERED 5 BEAMWIDTHS OFF-AXIS, IT WOULD PROVIDE PRECISION PERFORMANCE OVER A 9 BEAMWIDTH SECTOR. FOR EXTREMELY HIGH FREQUENCY SYSTEMS, LENSES MAY WELL REPLACE NETWORK MATRICES TO MINIMIZE NETWORK LOSSES AND COMPLEXITY.

SUMMARY

- FOR N ELEMENTS LITTLE DEGRADATION FOR GREATER THAN 0.65N BEAMS (ON AXIS)
- VALUE OF N PRINCIPALLY LIMITED BY SYSTEM COMPLEXITY AND/OR NETWORK LOSSES
- LOW FREQUENCY/SMALL N – NETWORK
- HIGH FREQUENCY/HIGH N – LENSES
- OFF AXIS CLUSTERS – A PRACTICAL COMPROMISE?
  (REDUCED COVERAGE FOR GIVEN N)

Figure 22

COMMENTS OF GENERAL INTEREST FROM QUESTIONS AND ANSWERS

Large Space Deployable Antenna Systems

1. Mesh Material for the Antenna Surface

The selection of a mesh in terms of material and weave must reflect the application for the antenna. The deployable configurations built to date have used woven dacron with copper plating or gold plating and wire meshes. The Lockheed Missiles and Space Company has made a documented compilation of data on meshes applicable to deployable antennas, copies are available upon request.

2. Bonding of Graphite Epoxy for Flexible Ribs

The construction of the furlable ribs has employed both riveted and secondary bonds for joining the edges of the graphite epoxy rib-halves. Secondary bonds have survived the high shear stresses associated with 60 furl-unfurl cycles without delamination.

ON SPACE DEPLOYABLE ANTENNAS AND ELECTRONICS: PLANS AND PROGRAMS

BY T. G. CAMPBELL, W. F. CROSWELL,
T. DEATON, AND B. DOBROTIN

T. G. CAMPBELL AND W. F. CROSWELL (NASA-LARC)

T. DEATON (NASA-MSFC)

B. DOBROTIN (JPL)

290

(Figure 1)

Presentation Outline: The presentation was divided into three parts as indicated in this vugraph. The first topic to be discussed is a development plan for large deployable reflectors.

PRESENTATION OUTLINE

I. A TECHNOLOGY DEVELOPMENT PLAN FOR LARGE DEPLOYABLE REFLECTORS (T. G. CAMPBELL)

II. ELECTROMAGNETIC ANALYSIS METHODS FOR LARGE REFLECTORS (W. F. CROSWELL)

III. A TECHNOLOGY PLAN FOR DEVELOPING ELECTRONIC SUBSYSTEMS FOR LARGE SPACE STRUCTURES (T. G. CAMPBELL, T. DEATON, AND B. DOBROTIN)

Figure 1

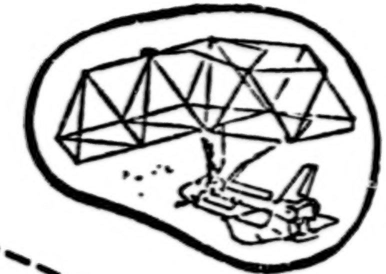
291

292

(Figure 2)

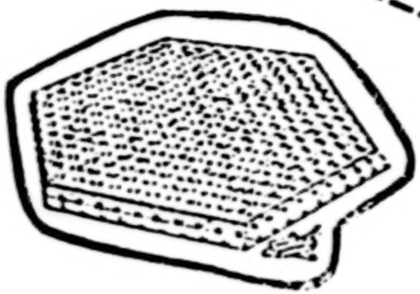
The LSST Project Office has established goals for the structural concepts needed for Space Shuttle missions in the 1985-2000 time period. As indicated in this vugraph, deployable reflectors 30-300 meters in diameter will be needed with a surface accuracy of several millimeters.

TECHNOLOGY FOR STRUCTURAL CONCEPTS NEEDED FOR SPACE SHUTTLE MISSIONS IN 1985-2000 TIME



ERECTABLE STRUCTURES

- 50-1000 METERS IN SIZE
- STRUCTURAL MASS ORDER OF MAGNITUDE LOWER
- EASILY ASSEMBLED FROM SHUTTLE
- EFFICIENTLY PACKAGED SO PAYLOAD NOT VOLUME LIMITED



DEPLOYABLE PLATFORMS

- PLANAR STRUCTURE
- 100-200 METERS
- CARRIED INTO ORBIT IN ONE SHUTTLE AND AUTOMATICALLY DEPLOYED

DEPLOYABLE REFLECTORS

- 30-300 METERS IN SIZE
- SURFACE MEASURED AND MAINTAINED TO ACCURACY OF FEW MILLIMETERS

Figure 2

293

294

(Figure 3)

This vugraph lists three basic questions that the LSST Project Office must address. In order for the LSST program to be a viable one for deployable reflectors, these questions must be resolved.

BASIC QUESTIONS THIS PROGRAM WILL ADDRESS

- o WHICH DEPLOYABLE REFLECTOR CONCEPTS WILL MEET THE REQUIREMENTS FOR LARGE SPACE STRUCTURES?
- o HOW LARGE CAN WE BUILD A DEPLOYABLE REFLECTOR?
- o WHAT CRITERIA WILL BE USED TO SELECT AN ERECTABLE DESIGN OVER A DEPLOYABLE REFLECTOR?

Figure 3

296

(Figure 4)

Typical technology goals for deployable reflector surfaces are listed in this vugraph for L and X-band applications. It can be seen that a 1 millimeter surface accuracy is to be achieved for 300 meter diameter reflectors.

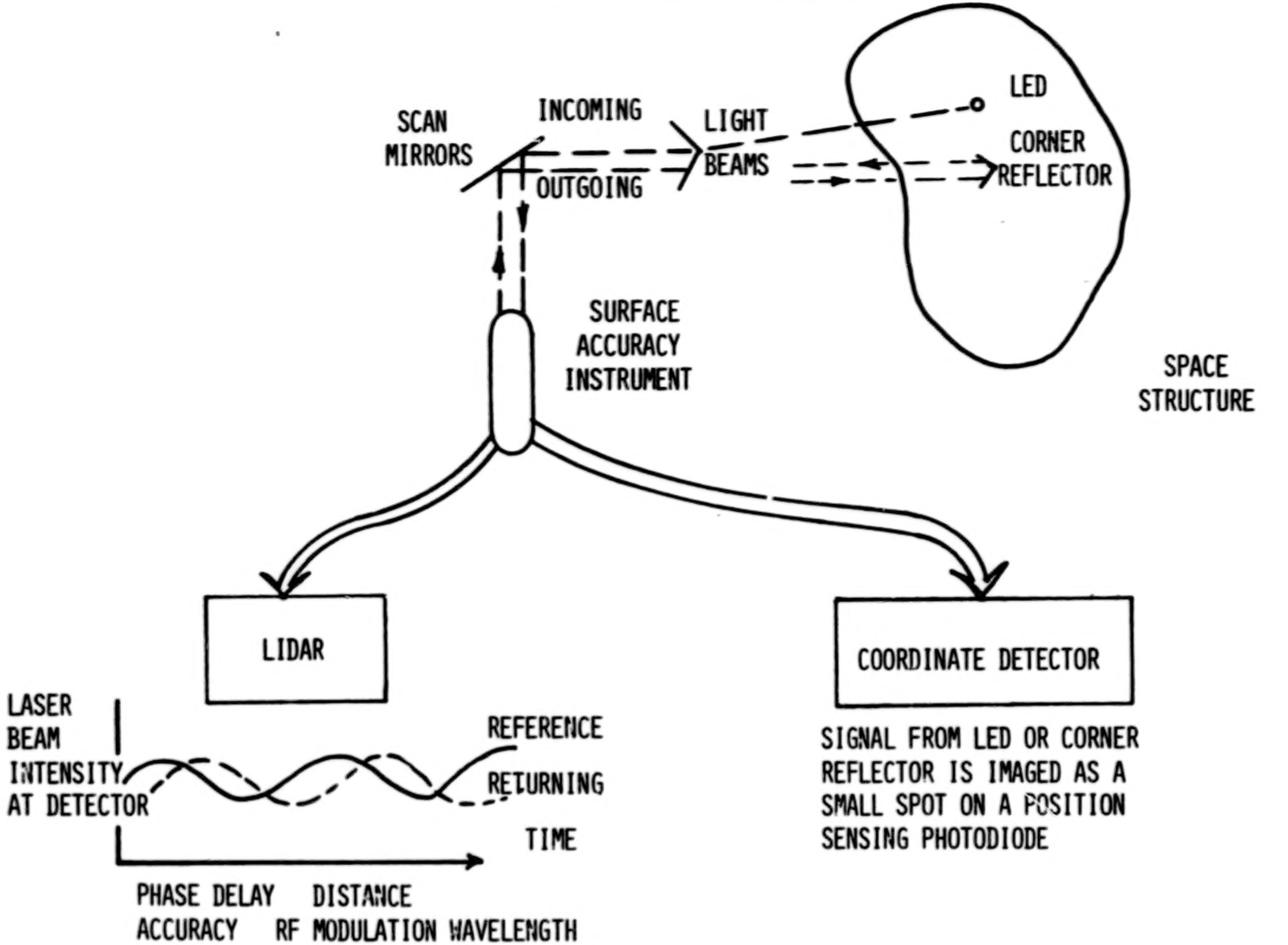
TYPICAL TECHNOLOGY GOALS FOR DEPLOYABLE
REFLECTOR SURFACES

	TODAY (S-BAND)	1985 (L TO X-BAND)	1995 (L TO X-BAND)
DIAMETER (METERS)	30	100	300
SURFACE ACCURACY δ/D RATIO	4.3×10^{-4} (δ = 13 MM)	$1.2 \times 10^{-4} \rightarrow 1.2 \times 10^{-5}$ (δ = 1 MM)	$4 \times 10^{-5} \rightarrow 4 \times 10^{-6}$ (δ = 1 MM)
GAIN DB	55	75	82
TRACKING RESOLUTION ARC SECONDS	41.3	37 → 6.2	12 → 2.1

Figure 4

(Figure 5)

Coupled with the need for large deployable reflectors is the need for a system that can measure 1 millimeter reflector surfaces commensurate with this range in diameters. Shown in this vugraph is a concept that has been proposed for this application.


SCAN
MIRRORS
INCOMING
OUTGOING
LIGHT
BEAMS
LED
CORNER
REFLECTOR
SURFACE
ACCURACY
INSTRUMENT
SPACE
STRUCTURE
LIDAR
COORDINATE DETECTOR
LASER
BEAM
INTENSITY
AT DETECTOR
REFERENCE
RETURNING
TIME
PHASE DELAY DISTANCE
ACCURACY RF MODULATION WAVELENGTH
SIGNAL FROM LED OR CORNER
REFLECTOR IS IMAGED AS A
SMALL SPOT ON A POSITION
SENSING PHOTODIODE

299

Figure 5

300

(Figure 6)

The technology tasks identified for the LSST project that are related to large deployable reflectors are shown in this vugraph. Four activities are listed and the development of a structural measurement sensor is included.

STRUCTURAL CONCEPTS - DEPLOYABLE REFLECTORS

1. CONCEPT DEVELOPMENT

   A. JUSTIFY AND SELECT CANDIDATE CONCEPTS
   B. DEVELOP DEPLOYMENT TECHNIQUES AND SURFACE PREDICTIONS
   C. SMALL SCALE COMPONENT TESTS
   D. TRADE-OFF STUDIES

2. GROUND TESTING AND TEST FACILITY REQUIREMENTS

   A. DEFINE REQUIREMENTS, PHILOSOPHY, AND LIMITATIONS
   B. IDENTIFY CANDIDATE FACILITIES AND NEEDED MODIFICATIONS

3. STRUCTURAL MEASUREMENT SENSOR DESIGN

   A. DEFINE MEASUREMENT CONCEPTS AND REQUIREMENTS
   B. BREADBOARD MOST PROMISING CONCEPTS

4. SURFACE ACCURACY SENSITIVITY EVALUATION

   A. DEVELOP SURFACE ACCURACY PERFORMANCE TECHNIQUES
   B. FORMULATE ANALYTICAL CHARACTERIZATION OF STRUCTURE

Figure 6

302

(Figure 7)

This vugraph identifies 10 candidate concepts that could be used for LSST deployable reflector applications. The purpose of the concept development task os LSST should be to review these concepts (and others not listed) and select a maximum of two for concept evaluation.

LARGE DEPLOYABLE REFLECTOR CONCEPTS

	CONCEPT	ORIGINATOR	ANTENNA TYPE	DIAMETER RANGE (M)
1.	EXPANDABLE TRUSS	GENERAL DYNAMICS	REFLECTOR	10-300
2.	WIRE WHEEL	GRUMMAN	PHASED ARRAY (REFLECTOR/LENS)	50-300
3.	HOOP/COLUMN	HARRIS	REFLECTOR	15-100
4.	ARTICULATED RIB	HARRIS	REFLECTOR	15-30
5.	CURVED ASTROMAST	HARRIS	REFLECTOR	15-<100
6.	RADIAL COLUMN	HARRIS	REFLECTOR	15-<100
7.	WRAP RIB	LOCKHEED	REFLECTOR	9-200
8.	POLYCONIC	LOCKHEED	REFLECTOR	30-300
9.	MAYPOLE	LOCKHEED	REFLECTOR	30-300
10.	SUNFLOWER	TRW	REFLECTOR	5

Figure 7

304

(Figure 8)

This vugraph outlines a possible technical plan for concept development of large deployable reflectors.

A TECHNICAL PLAN FOR LARGE DEPLOYABLE REFLECTORS

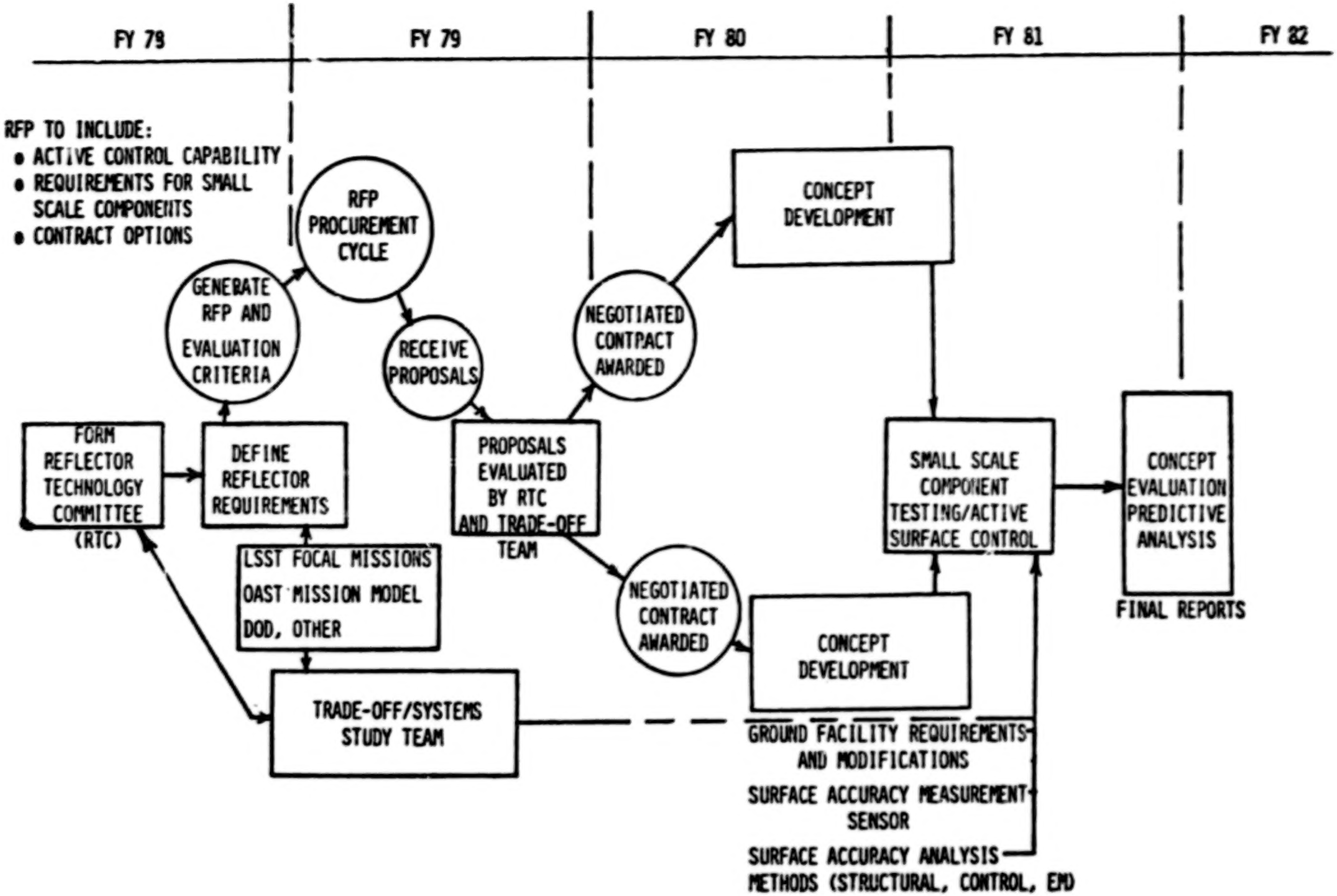


Figure 8

306

(Figure 9)

The probable results of this technology plan are listed in this vugraph and it must be recognized that active shape control will be included.

RESULTS OF TECHNOLOGY PLAN

PRODUCTS

- EVALUATION OF DEPLOYABLE REFLECTOR CONCEPT(S) THAT UTILIZE ACTIVE SHAPE CONTROL
- DEVELOPMENT AND VERIFICATION OF STRUCTURAL ANALYSIS METHODS FOR LARGE DEPLOYABLE REFLECTORS
- DEVELOPMENT AND VERIFICATION OF ELECTROMAGNETIC ANALYSIS METHODS FOR LARGE DEPLOYABLE REFLECTORS
- DEVELOPMENT OF A SURFACE ACCURACY MEASUREMENT SYSTEM AND PERFORMANCE VERIFICATION

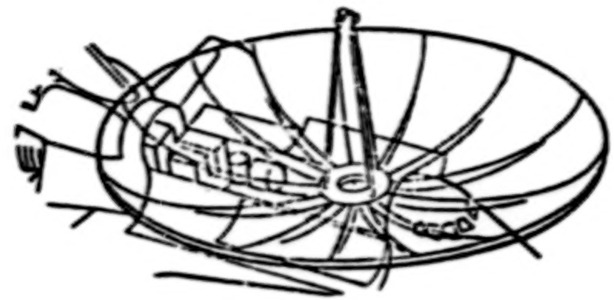
Figure 9

308

(Figure 10)

As mentioned previously, large deployable reflector concepts will be investigated through the LSST program. Included in these investigations must be the development of analysis methods that will be used to predict the radio frequency performance of these large reflectors.

DEVELOPMENT OF ANALYSIS METHODS FOR LARGE DEPLOYABLE REFLECTORS



DEPLOYABLE REFLECTORS

- 30 - 300 METERS IN SIZE
- SURFACE MEASURED AND MAINTAINED TO ACCURACY OF FEW MILLIMETERS

ANALYSIS METHODS

- PREDICT RADIO FREQUENCY PERFORMANCE FOR LARGE REFLECTORS

Figure 10

(Figure 11)

This vugraph compares an analysis method using sectorial approximation with Ruze's formula. It can be seen that the surface accuracy requirements for a given gain degradation are not as stringent as that predicted by Ruze's formula.

ANTENNA SURFACE ACCURACY EFFECTS

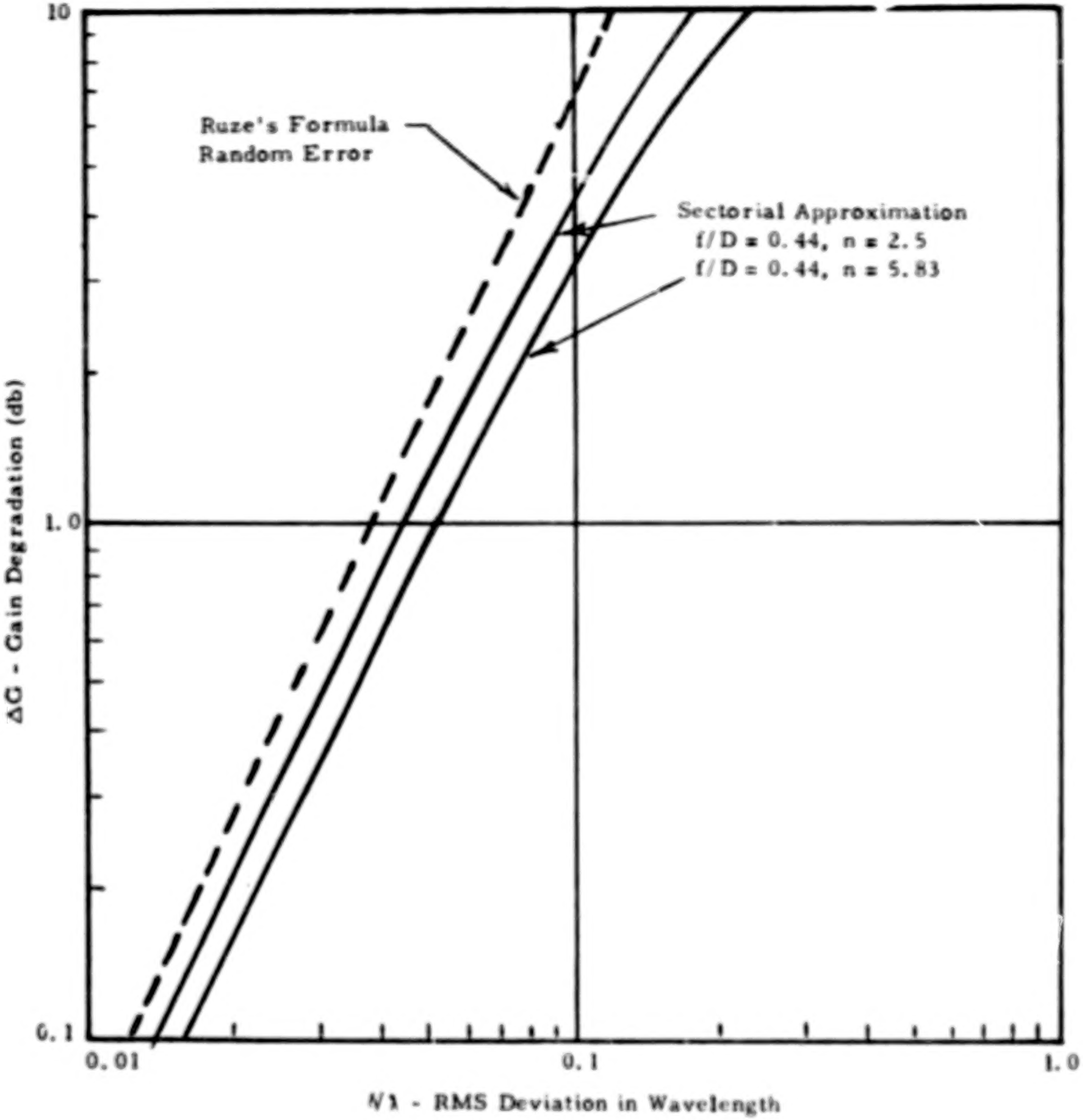
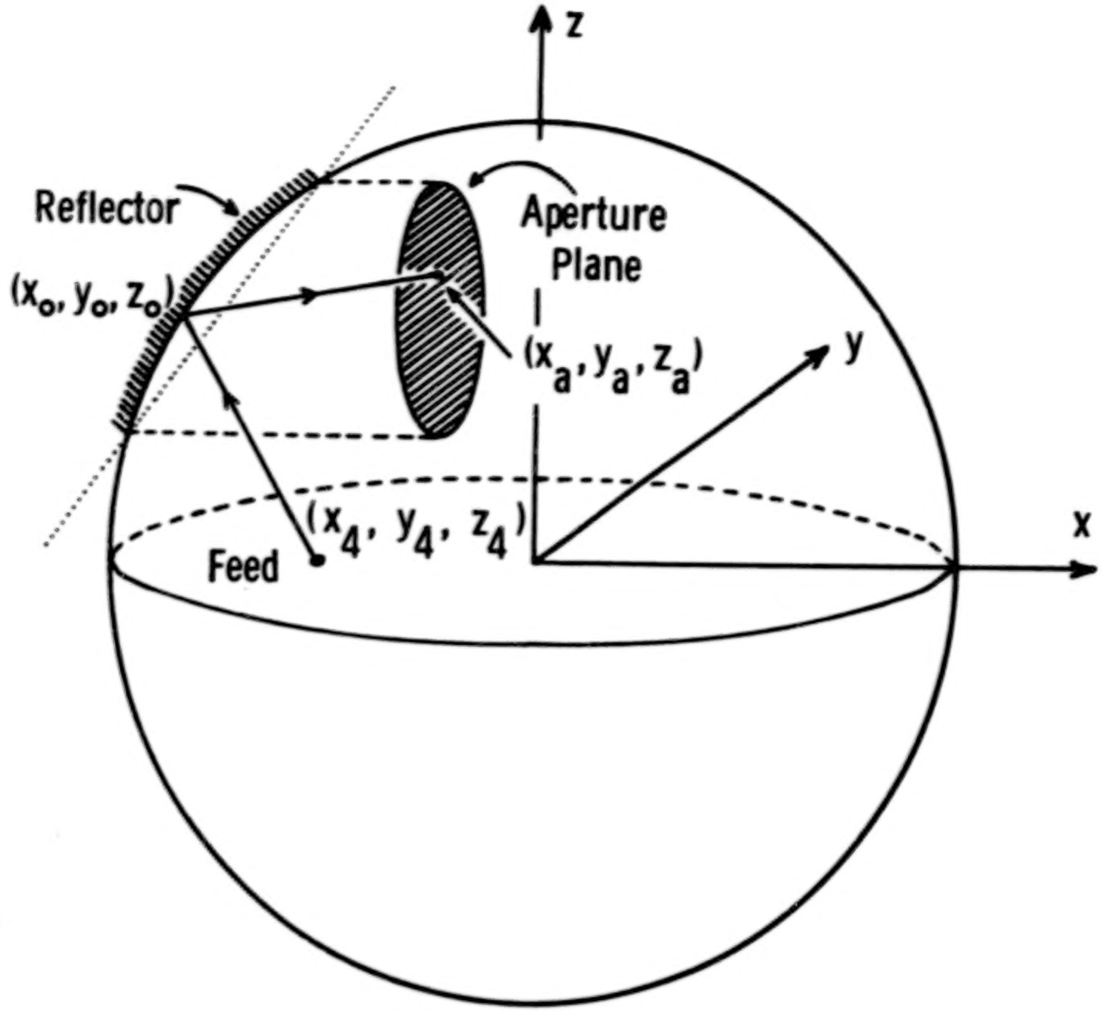


Figure 11

(Figure 12-Figure 16)

These vugraphs describe the electromagnetic analysis method whereby the field points are projected onto an aperture plane. The data points in the aperture plane are quantized to produce contours of constant phase and amplitude. The far field patterns are then calculated using reduced computer storage.

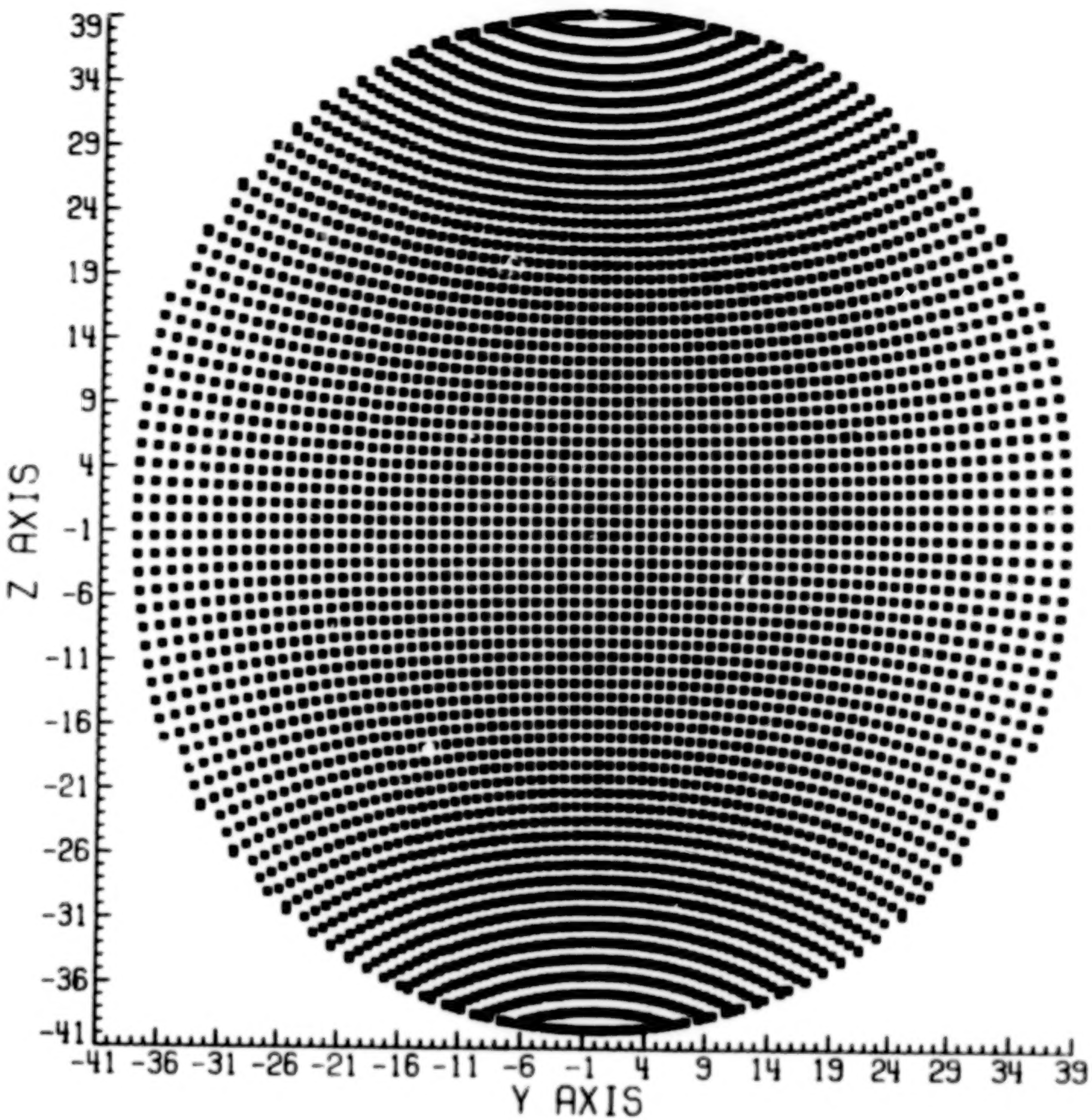


Antenna Geometry

(Figure 12)

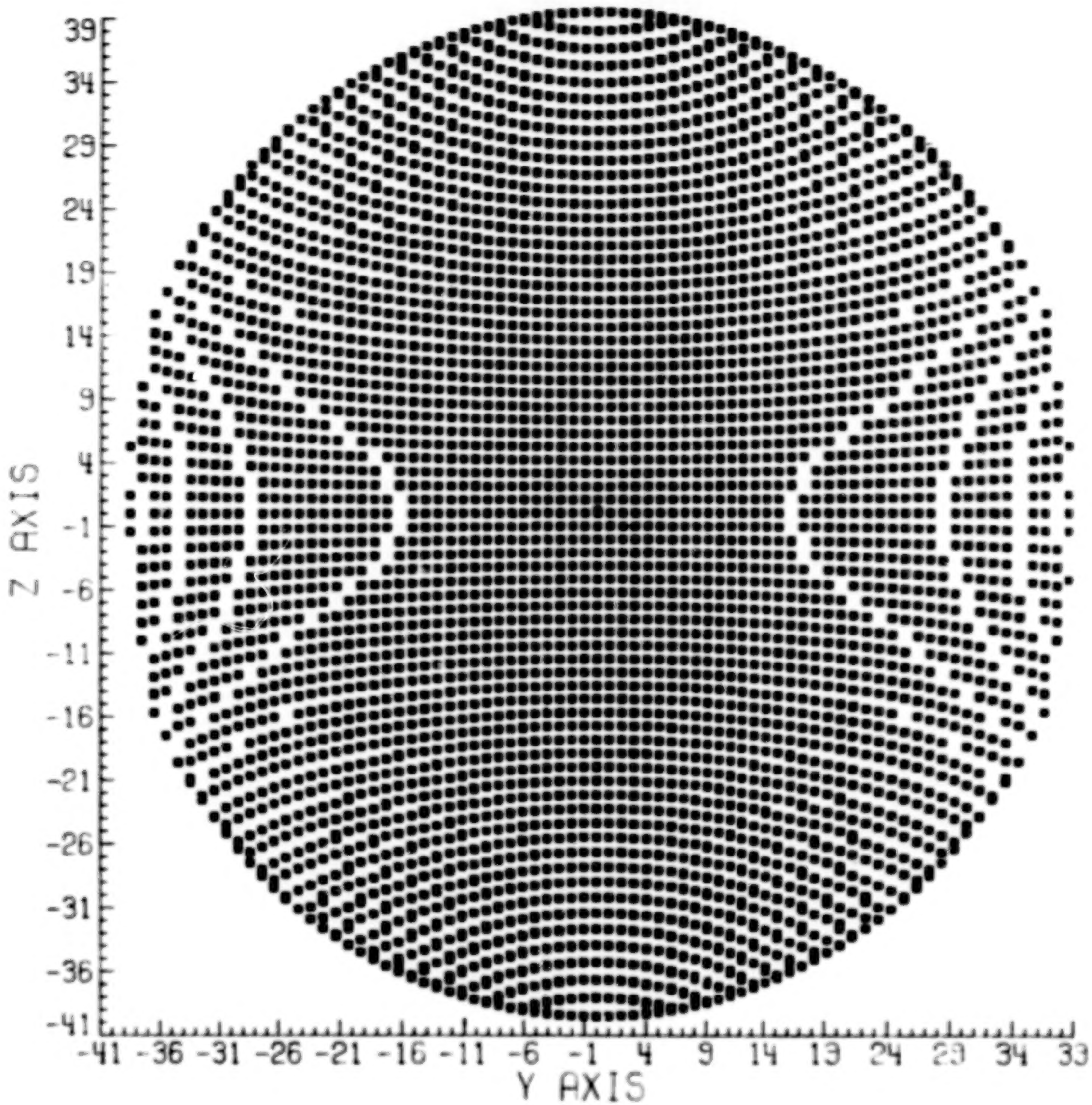
313

314



Data Point Locations in the Aperture Plane

Figure 13

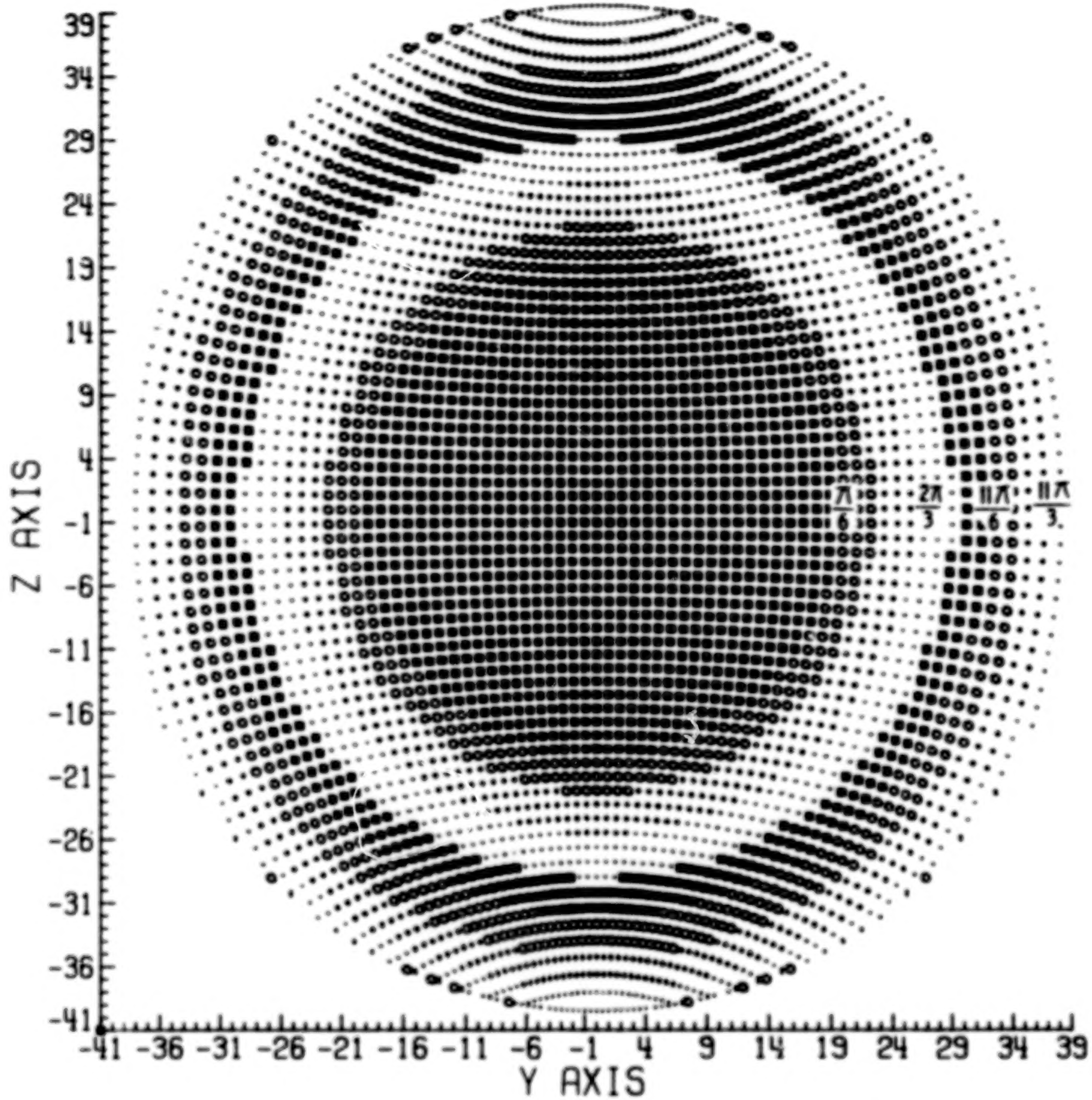


Quantized Data Point Locations in the Aperture Plane

Figure 14

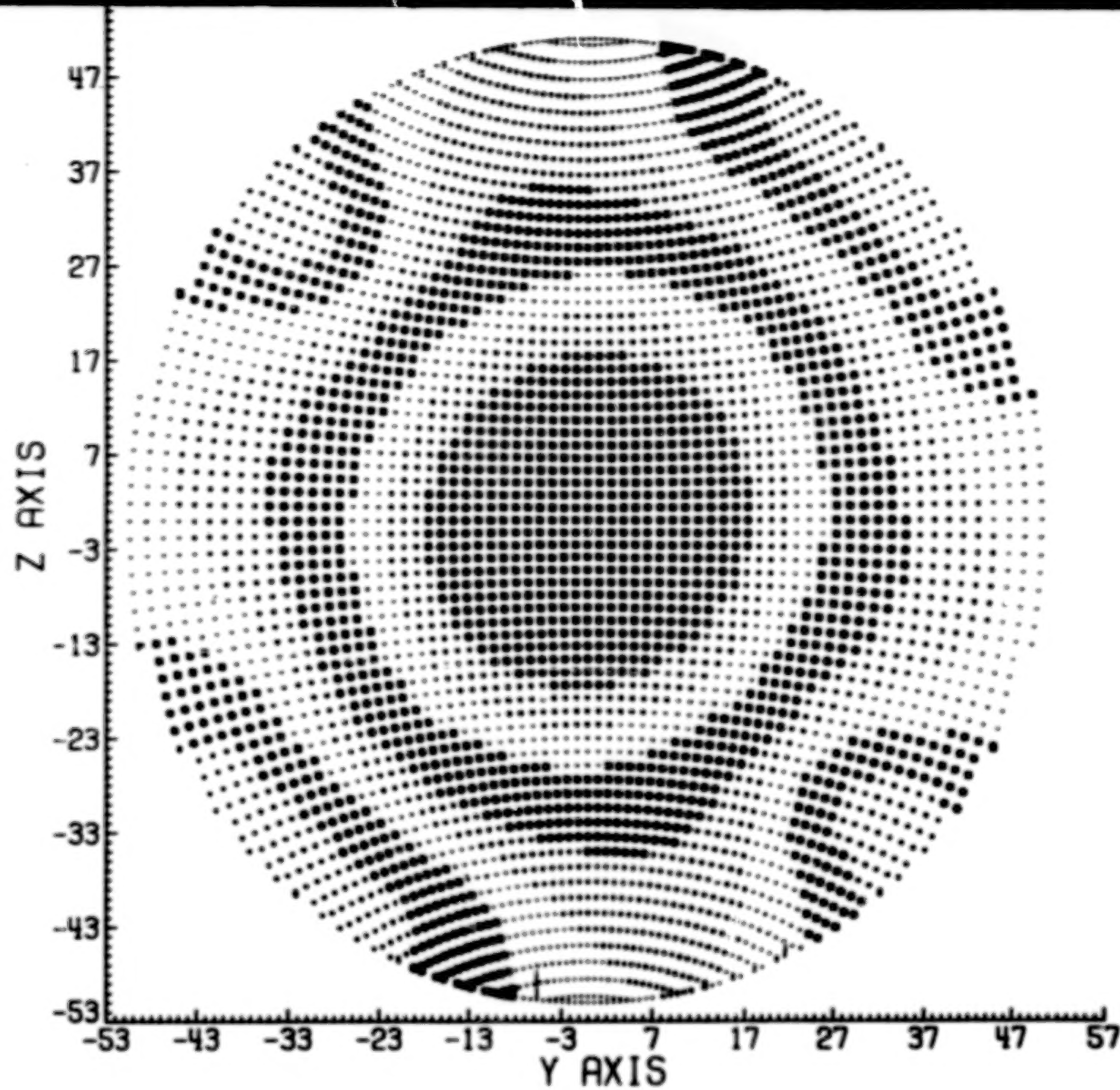
315

316



Contours of Constant Phase in the Aperture Plane

Figure 15



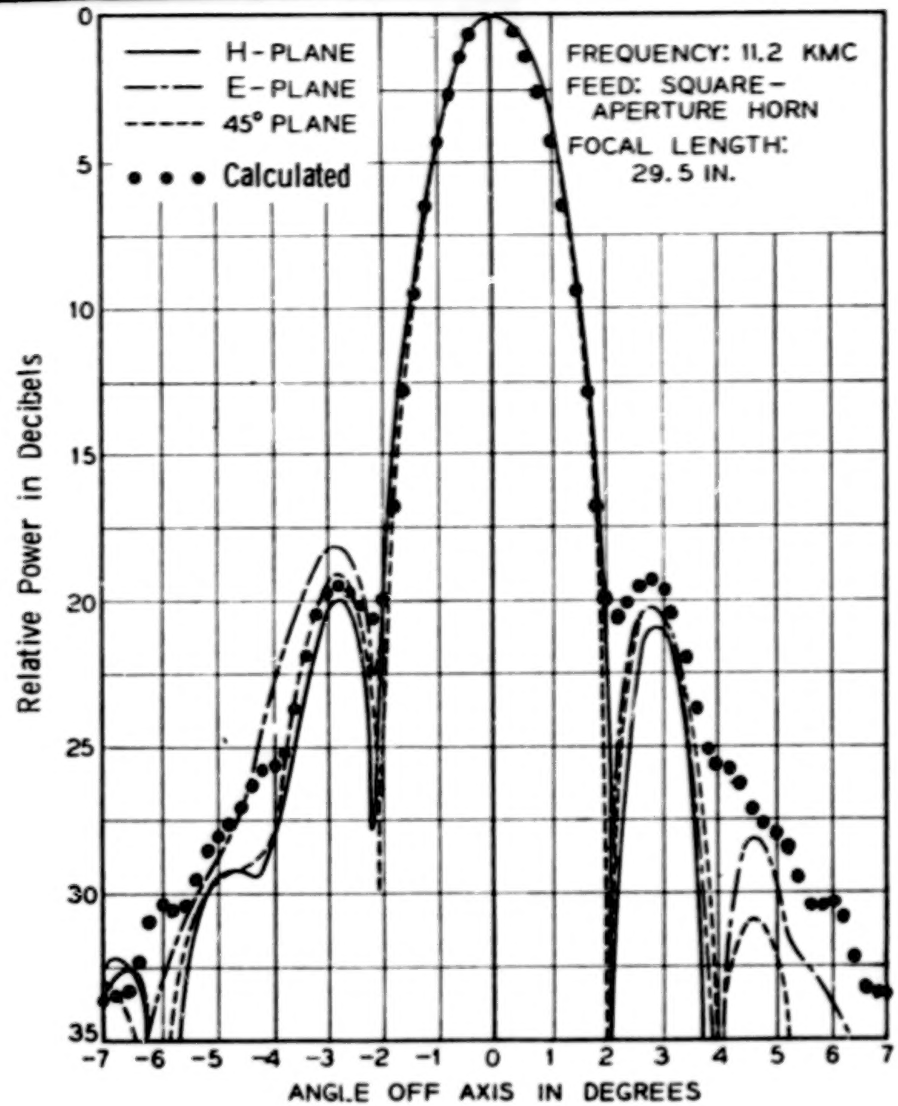
Contours of Constant Amplitude in the Aperture Plane

Figure 16

317

(Figure 17)

The radiation patterns of a 10-foot diameter spherical reflector were computed and compared with measured data in this vugraph. It can be seen that good agreement was observed. This analysis plan will be used in LSST.



Radiation Patterns of 10-foot Spherical Reflector

(Measurements by Li, 1959)

Figure 17

319

320

(Figure 18)

This vugraph lists the tasks for the electromagnetic analysis activity for LSST. This plan extends through FY 82.

TASKS	FY 1978	FY 1979	FY 1980	FY 1981	FY 1982

a) **Assess State of-the-Art**

- Complete review of existing reflector analysis methods.
- Complete review of existing techniques for analysis of large arrays.

b) **Develop Modeling Techniques**

- Develop physical optics-aperture integration method for segmented reflectors (segments of known surfaces).
- Develop ray optical (G. T. D.) analysis of segmented reflectors.
- Develop analysis of segmented distorted reflectors.
- Construct and measure rf properties of antenna models to verify basic analysis techniques and methods.
- Develop analysis of random and deterministic errors for large arrays
- Develop techniques for analysis of distorted, segmented arrays.

c) **Define Discipline Interface Requirements**

- Determine the antenna geometry requirements for in orbit prediction and control of antenna performance.

d) **Develop New Analysis Techniques**

- Complete analysis and computer programs for large reflectors of known shape.
- Develop analysis and computer programs for large segmented reflectors.
- Develop analysis and computer programs for large segmented arrays.

Figure 18

322

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A TECHNOLOGY PLAN FOR

DEVELOPING ELECTRONIC SUBSYSTEMS

FOR LARGE SPACE STRUCTURES

323

324

(Figure 19)

This electronics plan must address the basic questions listed in this vugraph.

BASIC QUESTIONS THE ELECTRONICS PROGRAM WILL ADDRESS:

- HOW WILL ELECTRICAL POWER, DATA, AND COMMAND SIGNALS BE TRANSMITTED THROUGH A LARGE SPACE STRUCTURE?
- WHAT TYPE OF GROUNDING (FAULT CURRENT BONDING) SYSTEM WILL BE USED IN A LARGE SPACE STRUCTURE?
- CAN FIBER OPTICAL TECHNIQUES BE USED IN LARGE SPACE STRUCTURES?

Figure 19

325

CONTENTS

VOLUME I

326

(Figure 20)

The objectives of the electronics plan are listed in this vugraph.

OBJECTIVES

- TO DEVELOP TECHNOLOGY FOR DISTRIBUTED ELECTRICAL SUBSYSTEMS
- TO IDENTIFY FUTURE REQUIREMENTS WHICH EXCEED STATE-OF-THE-ART TECHNIQUES AND COMPONENTS
- TO INITIATE THE DEVELOPMENT OF NEW TECHNIQUES FOR DATA AND POWER MANAGEMENT/DISTRIBUTION
- TO INITIATE THE ADVANCEMENT OF COMPONENTS FOR DATA, POWER, AND CONTROL SUBSYSTEMS

Figure 20

328

(Figure 21)

The LSST Project Office has listed the technology activities for electronics that appear to be within the funding guidelines. These activities are listed in this vugraph. It can be seen that the activities for measurements and control devices have been overguidelined from the present LSST budget requests.

1. ELECTRONIC SYSTEM REQUIREMENTS DEVELOPMENT
   A. MISSION REQUIREMENTS DEFINITION
   B. MODEL DEVELOPMENT

2. DATA SYSTEM CONCEPTS
   A. DATA MANAGEMENT TECHNIQUES
   B. SIGNAL CONDITIONING

3. POWER SYSTEM INTEGRATION
   A. POWER MANAGEMENT TECHNIQUES
   B. POWER CONDITIONING
   C. POWER EQUIPMENT INTEGRATION

4. DATA AND POWER DISTRIBUTION
   A. CONNECTORS/CABLES
   B. RF DISTRIBUTION TECHNIQUES
   C. FIBER OPTICS DISTRIBUTION TECHNIQUES
   D. GROUND/BONDING TECHNIQUES

5. MEASUREMENTS*

6. CONTROL DEVICES*

*OVERGUIDELINED

Figure 21

329

(Figure 22)

This vugraph presents the schedule for the electronics activities for LSST. The success of the electronics tasks will depend upon the definition of mission requirements and the subsequent "model" development. The electronics subsystem model will be computerized so that network management can be conducted.

ELECTRONIC SYSTEMS ACTIVITIES SCHEDULE
(LATEST START DATES)

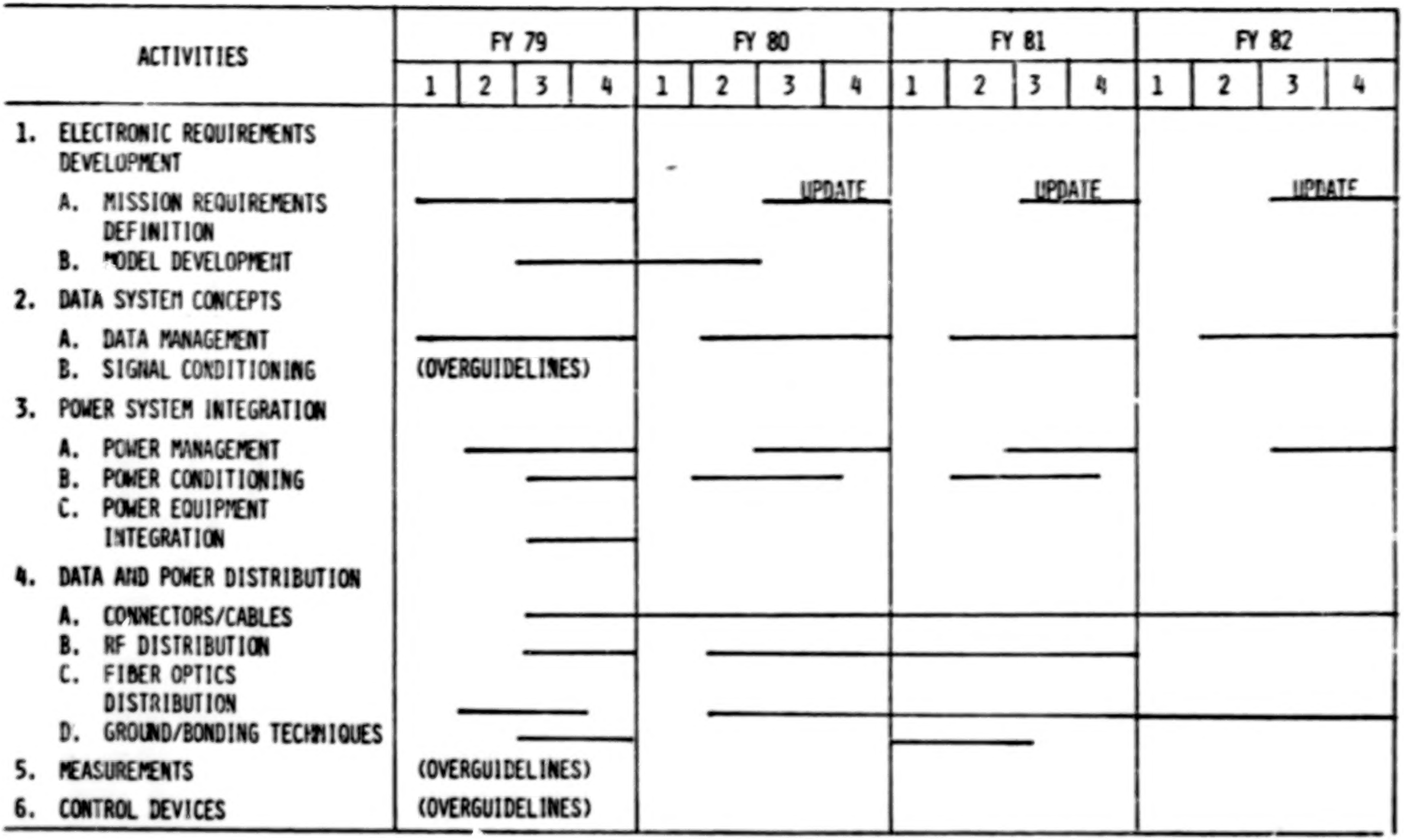

Figure 22

332

(Figure 23)

This vugraph presents the relationship of all the electronic activities. The development of new components will be emphasized through this activity.

ELECTRONIC ACTIVITIES RELATIONSHIPS

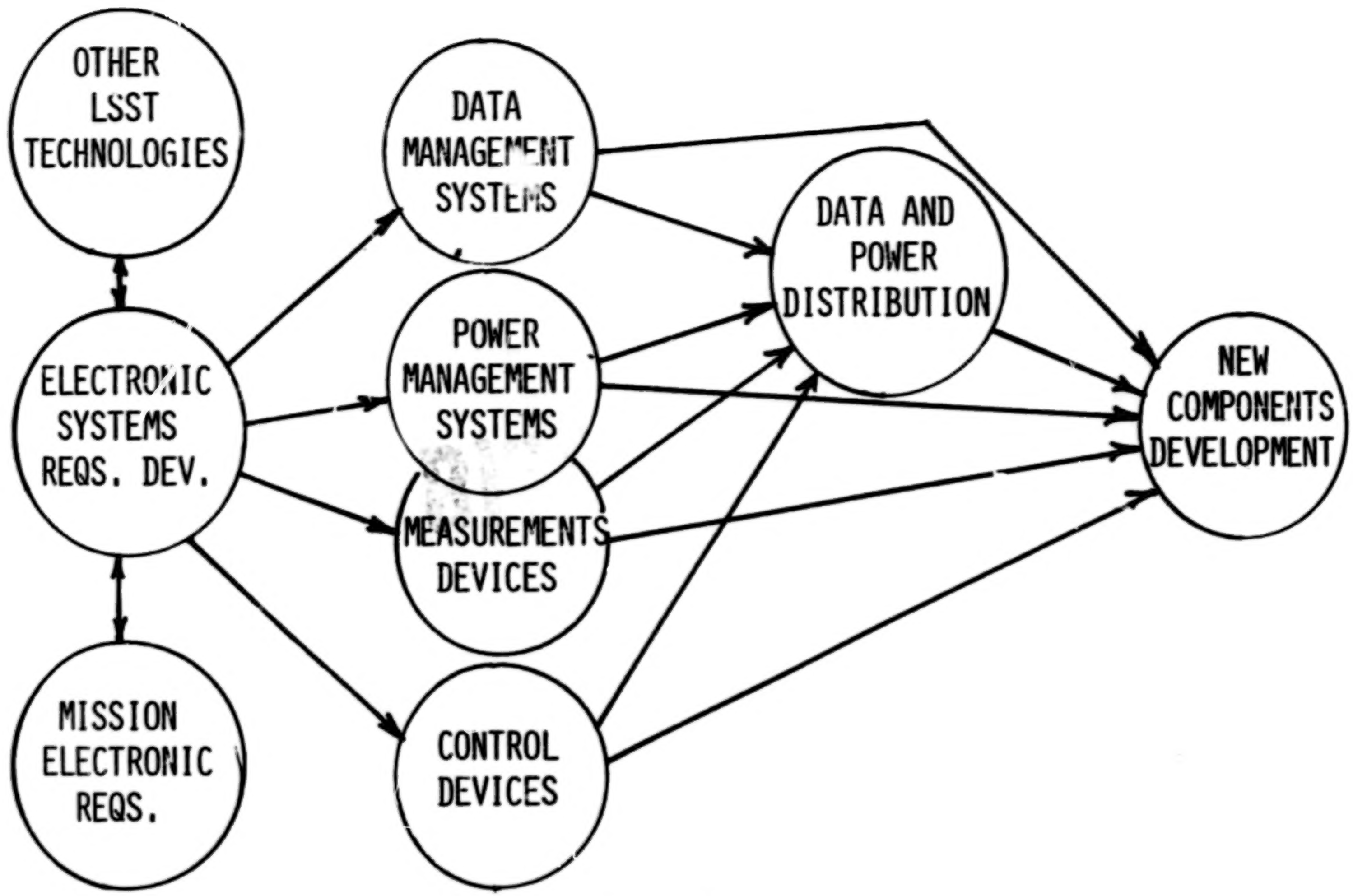

Figure 23

334

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APPLICATION OF GEO-TRUSS ERECTABLE ANTENNA

1985 - 2000 SYSTEMS

John A. Fager

Presented at

GOVERNMENT/INDUSTRY SEMINAR

ON

LARGE SPACE SYSTEMS TECHNOLOGY

NASA Langley Research Center

January 17-19, 1978

APPLICATION OF GEO-TRUSS ANTENNA CONCEPT TO YEAR 1985 - 2000 SYSTEMS (Figure 1)

John A. Fager
Program Manager of Special Projects
General Dynamics/Convair Division

This brief paper a[illegible]s to rough out three potential systems of the 1985 - 2000 period for the purpose of sizing realistically large structural systems — primarily antennas and accompanying support structures. Since the large volume mass moment of inertia of the antennas predominate, they become the spacecraft and the other subsystems attach to it. The geo-truss concept provides a natural structural element to use in the deployment or fabrication of these large systems.

In general, reflector systems were selected over lenses or phased arrays due to their economy, simplicity, and weight advantages. Frequency, beamwidth and gain requirements determine the antenna size and surface contour control.

Three systems are conceptually proposed:

- Direct TV Broadcast to half-time zone, Alaska and Hawaii
- Deep Space Communication Satellite
- Coastal Water Surveillance Radar Satellite

Due to their high cost and assembly requirements it is assumed, independent of orbit, that a docking facility required during fabrication would allow refurbishment of the electronics and power systems to increase system life.

APPLICATION OF GEO-TRUSS ERECTABLE ANTENNA
1985 - 2000 SYSTEMS

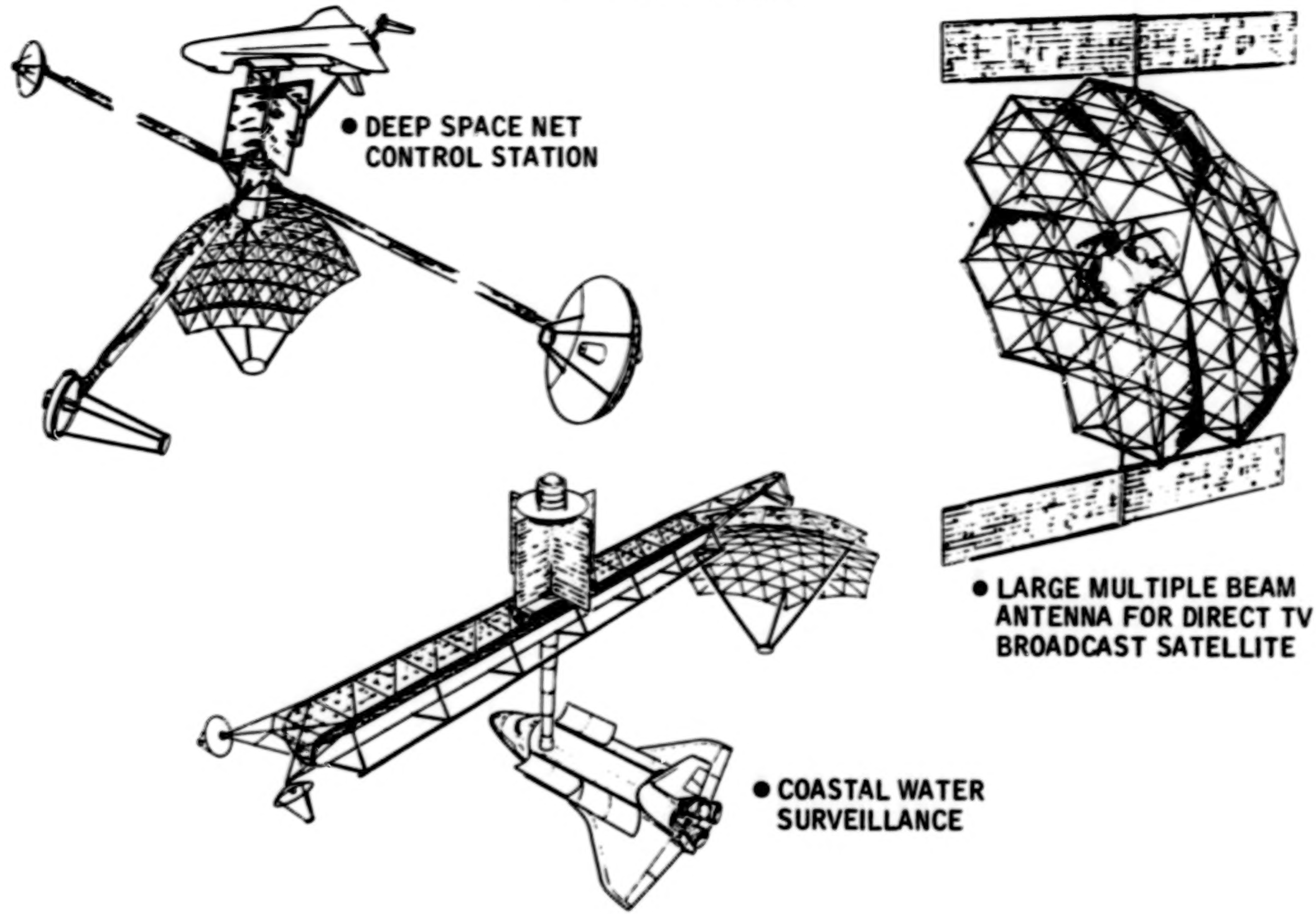


Figure 1

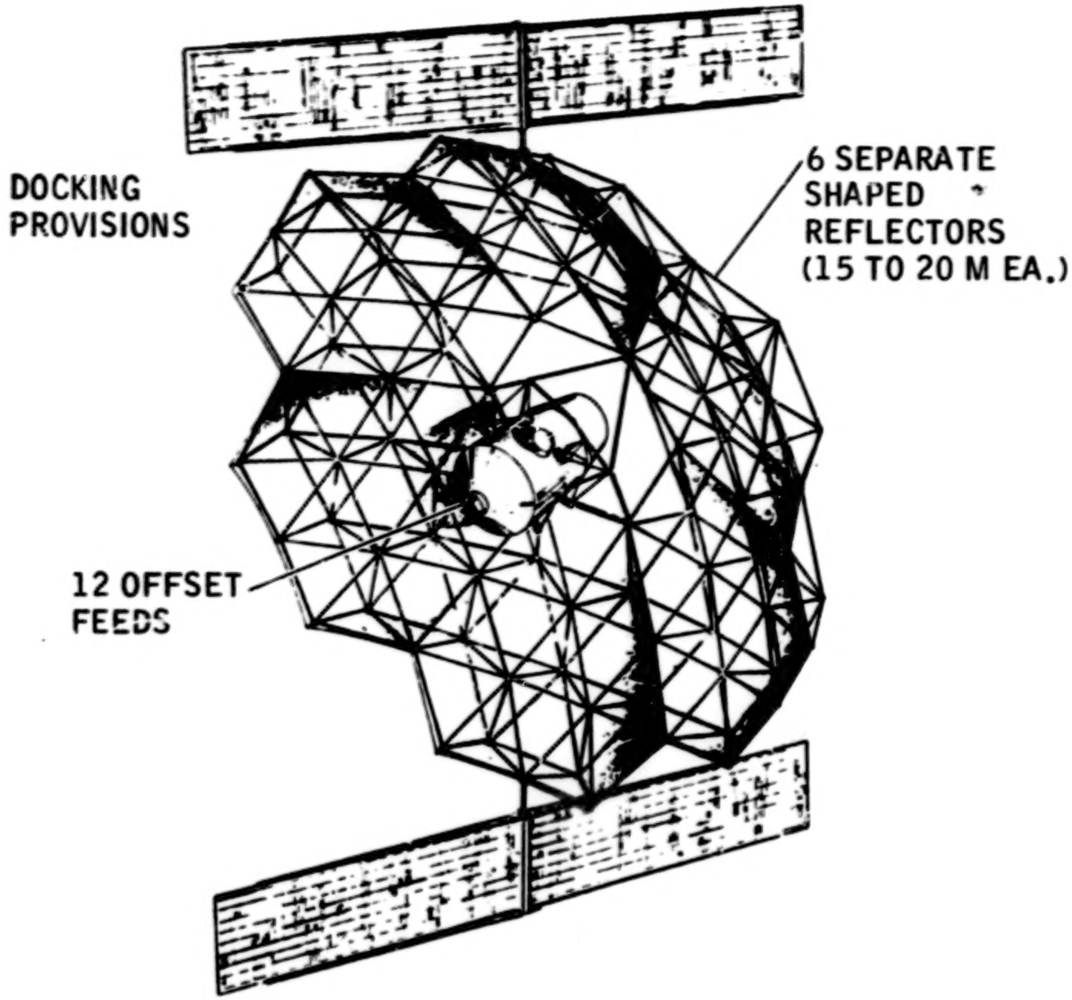
338

DIRECT TV BROADCAST SATELLITE (Figure 2)

Direct to home television service requires radiated power in the 80 dbw range. In order to reduce satellite power and home receiver cost, high gain systems are needed.

This large geodetic-truss has six shaped reflectors built into a common structure. An offset, zero blockage and feed is used to provide two beams to each time zone. A dock cone at the pressurized electronic/feed element will be used during assembly and checkout. It will also be used for replacement of the high power transmitters and other components. Size is a function of the broadcast frequency.

DIRECT TV BROADCAST SATELLITE CHARACTERISTICS



- LARGE STRUCTURE (45 TO 65 METER DIAMETER)
- SIX REFLECTOR SYSTEM INTEGRAL IN ONE STRUCTURE
- TWO BEAMS PER REFLECTOR
- MINIMUM SIDELOBES
- MAXIMUM GAIN
- MANNED SUPPORT FOR REFURBISH/MAINTENANCE IN PRESSURIZED FEED/ TRANSMITTER SECTION
- OFFSET FEEDS (MIN. BLOCKAGE)
- TRANSMITTER PACKAGE ADJACENT TO FEEDS
- SOLAR CELLS MOUNTED DIRECTLY TO GEO-TRUSS ANTENNA

Figure 2

ANTENNA SIZE REQUIREMENTS (Figure 3)

This chart shows the general coverage ranges as a function of frequency, beamwidth and antenna diameter. Pointing accuracy is usually 1/10 of beamwidth. Two degrees would cover a time zone in the U.S. while 1/8 degree would intersect a 50 mile zone.

Proposed TV Broadcast Satellite system with a 1-degree beamwidth would provide north and south coverage for each time zone in the U.S. At 640 MHz, a 100-foot (30.48 meter) antenna or 28 feet (8.53 meters) at 2.5 GHz would produce the required 1-degree beamwidth.

ANTENNA SIZE REQUIREMENTS FOR VARYING COVERAGE & FREQUENCY VARIATIONS

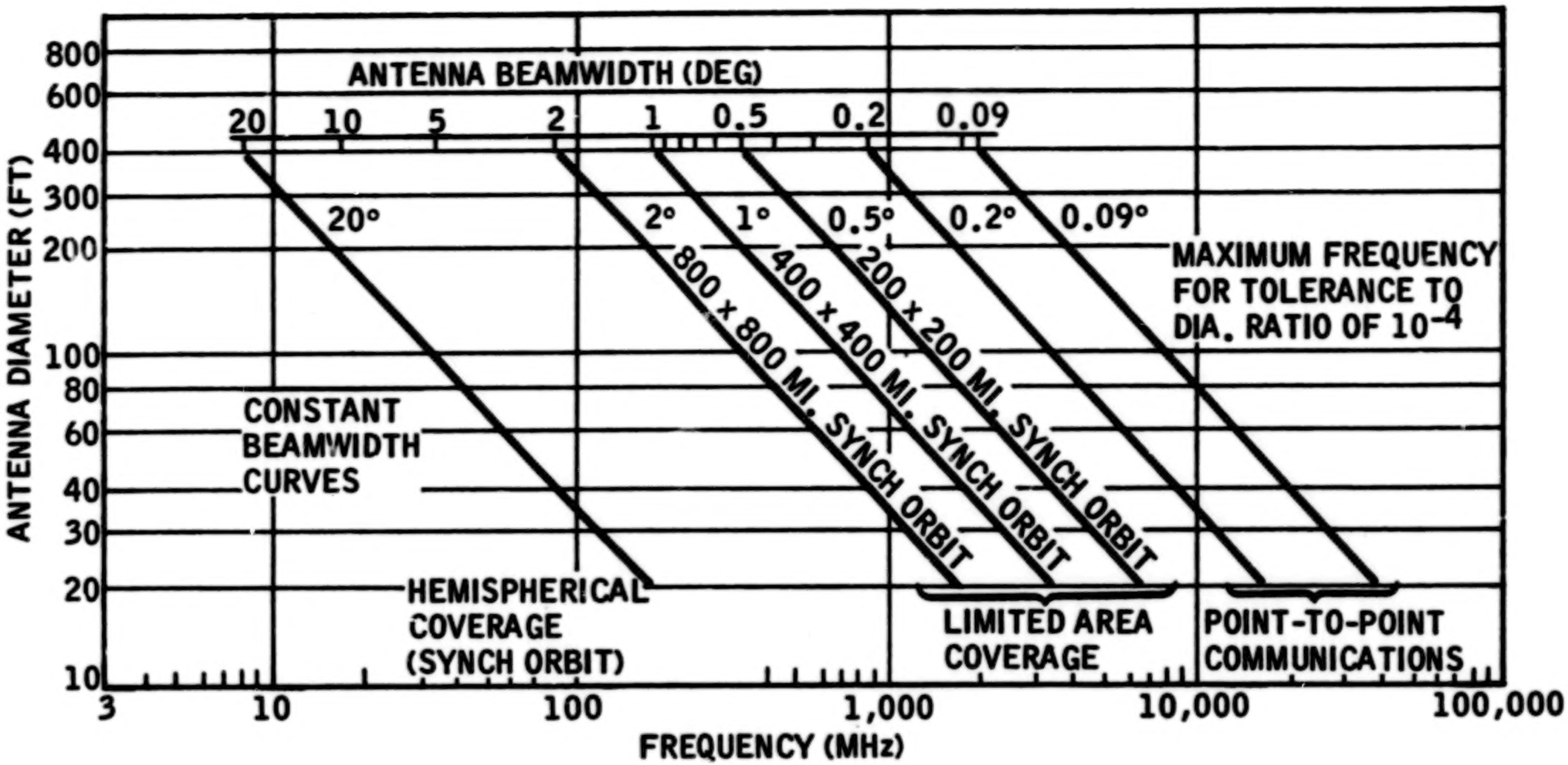


Figure 3

341

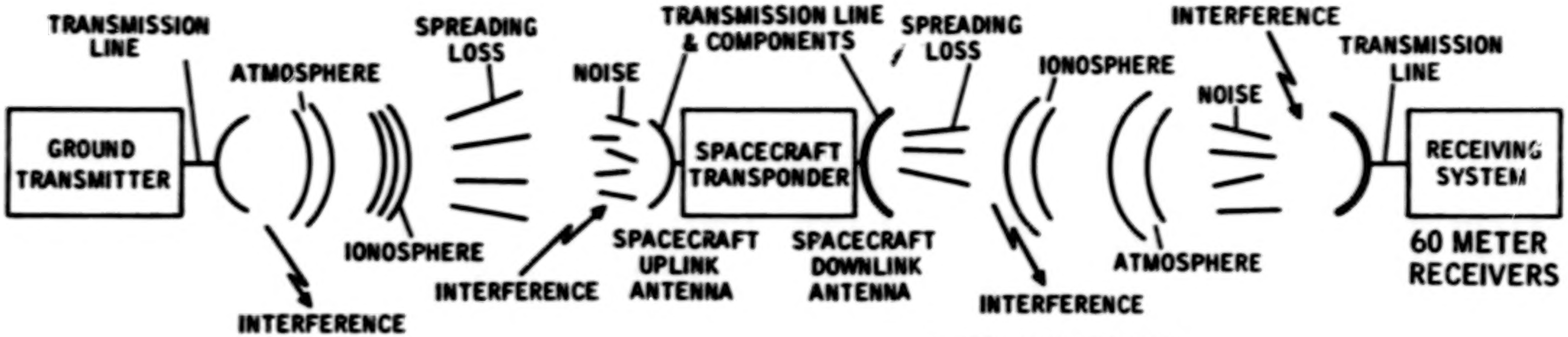
342

(Figure 4)

With 60 million potential TV receivers for a direct broadcast satellite in the U.S., assuming the goal is to provide the service at the lowest cost, considerable funds can be expended in the satellite to reduce the cost of the ground receiving system. Link tradeoffs show that increasing gain in the space antenna, large diameters will reduce cost. For example, use of an Intelsat IV type satellite for the Alaska communication system costs four times that of a large high gain system that could transmit directly to homes.

REASON LARGE ANTENNAS ARE REQUIRED FOR DIRECT TV BROADCAST SATELLITE

LINK CHARACTERISTICS:



F = FREQUENCY

COST & TYPE VARIABLE WITH F & MODULATION	ANTENNA SIZE $\propto \frac{1}{F}$	ABSORPTION & SCATTERING $\propto F$	LOSS $\propto F^2$	ANTENNA SIZE $\propto \frac{1}{F}$	TRANSPONDER GAIN (A) NOISE TEMP	ANTENNA SIZE $\propto \frac{1}{F}$ $\propto$ USER COVERAGE	LOSS $\propto F^2$	ABSORPTION SCATTERING POLARIZATION ROTATION	NOISE TEMP $\propto \frac{\text{COST}}{F}$	ANTENNA SIZE & QUALITY $\propto \frac{\text{COST}}{F}$	COST & TYPE VARIABLE WITH F & MODULATION

- INCREASING THE RADIATED POWER OF DOWNLINK ANTENNA WILL ALLOW REDUCED SIZE/ QUALITY OF HOME RECEIVING SYSTEM.
- A DOLLAR SAVING IN HOME RECEIVING SYSTEM COULD BE BALANCED BY $60M IN SATELLITE.

Figure 4

344

(Figure 5)

The relationship of the ground station, quality of TV service, and satellite power can be seen on the right hand chart. Coverage area determined by beamwidth sets the gain of the antenna for a given frequency. Based on the antenna gain, a compatible satellite average transmitter power can be derived. Frequency variations can be developed by comparing the 870 MHz system to the next 2.5 GHz chart. (2500 MHz)

GEOSTATIONARY TV BROADCAST NOMOGRAM (870 MHz)

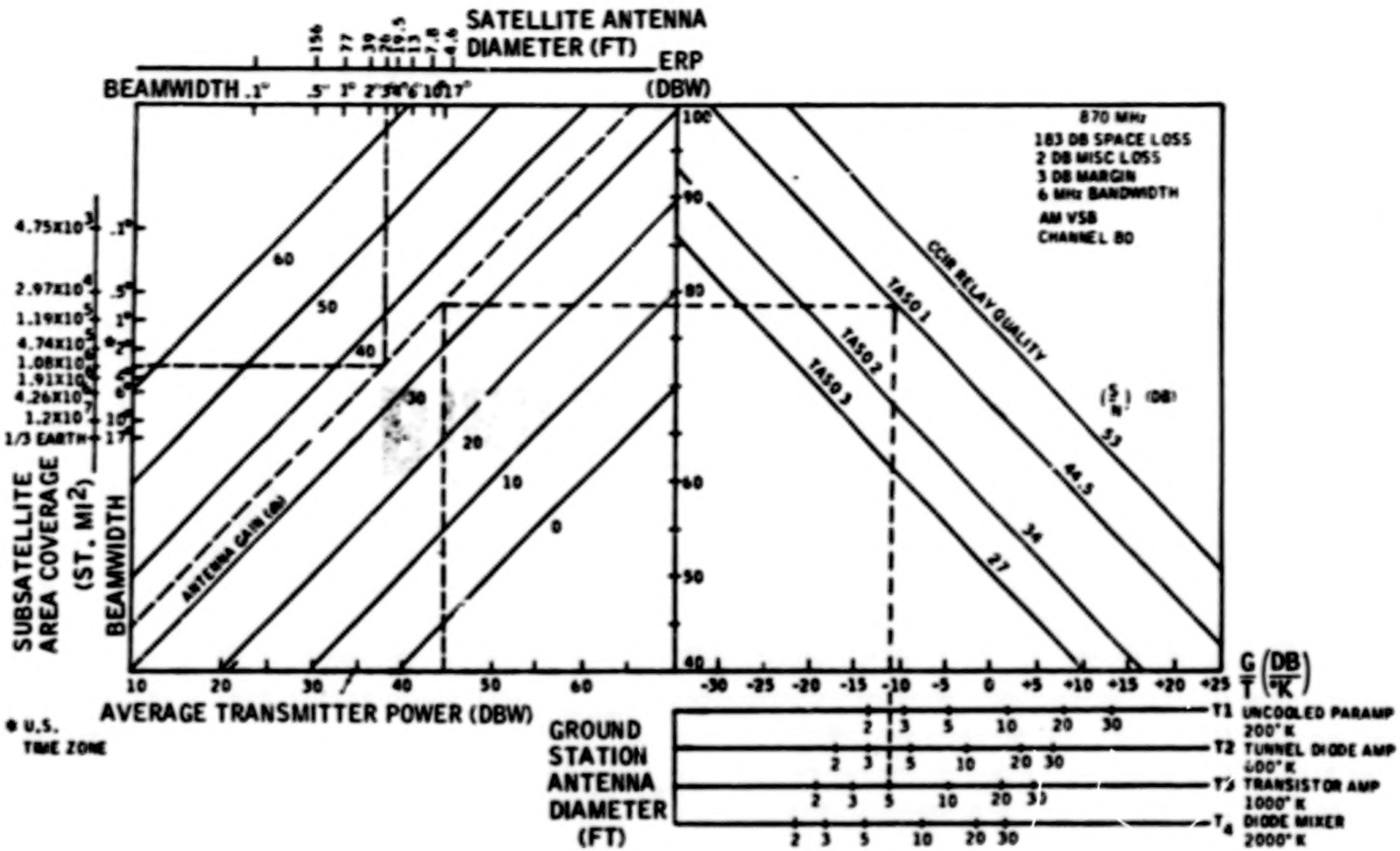


Figure 5

346

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GEOSTATIONARY TV BROADCAST NOMOGRAM (2,500 MHz)

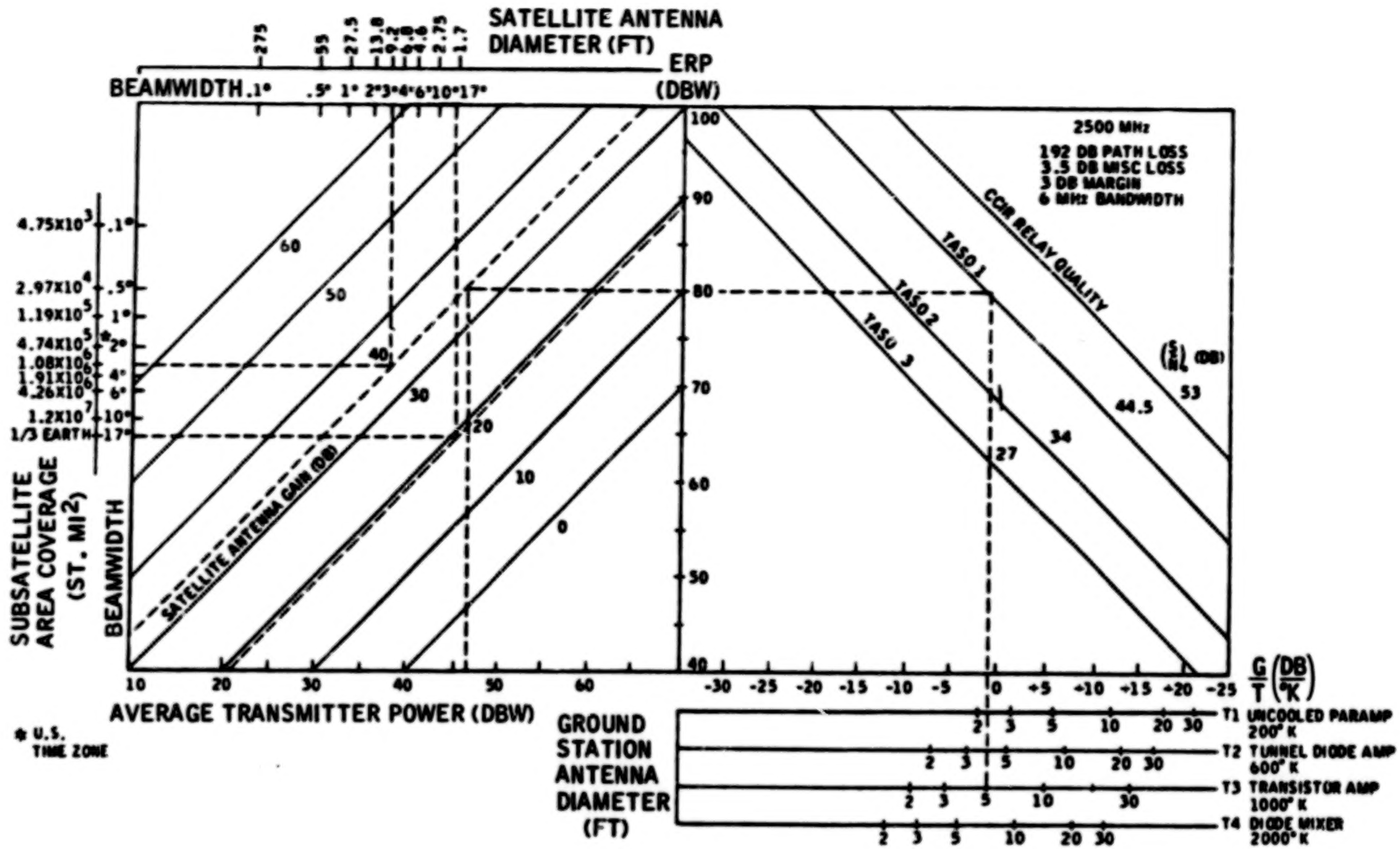


Figure 6

348

(Figure 7)

The three key parameters that control the radiated power (ERP) of a satellite are the solar cell power, transmitter and antenna diameter. While the various efficiency of these components can change, based on the subsystem selected, these charts show the trends that can be expected. For a given ERP, there is an optimum antenna diameter for the system assuming beamwidth and regulatory requirements are met.

SATELLITE ANTENNA SIZE TRADEOFF FOR

TVBS RADIATED POWER LEVEL

(ASSUMING BEAMWIDTH ADEQUATE)

- COST & WEIGHT INCLUDE ANTENNA, SOLAR CELLS & TRANSMITTER SYSTEM
- OPTIMIZATION CHANGES WITH FREQUENCY

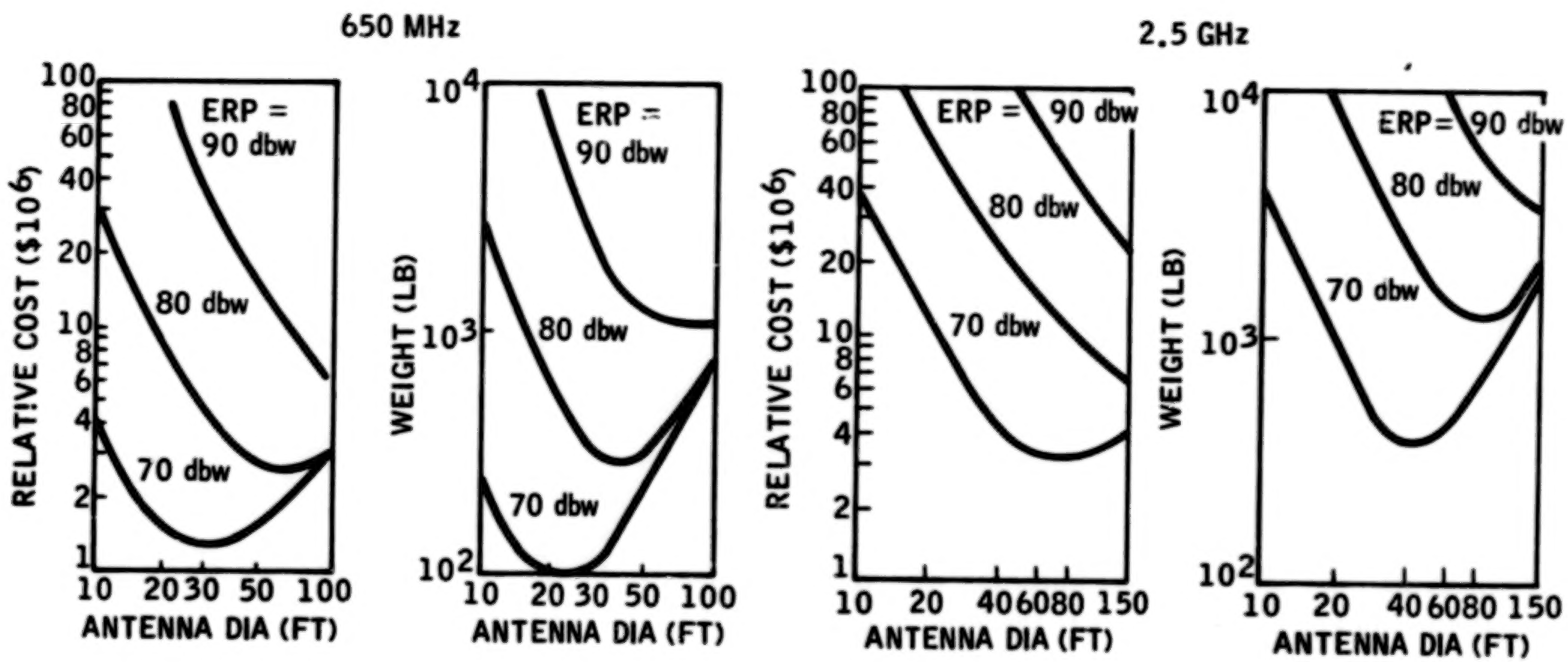


Figure 7

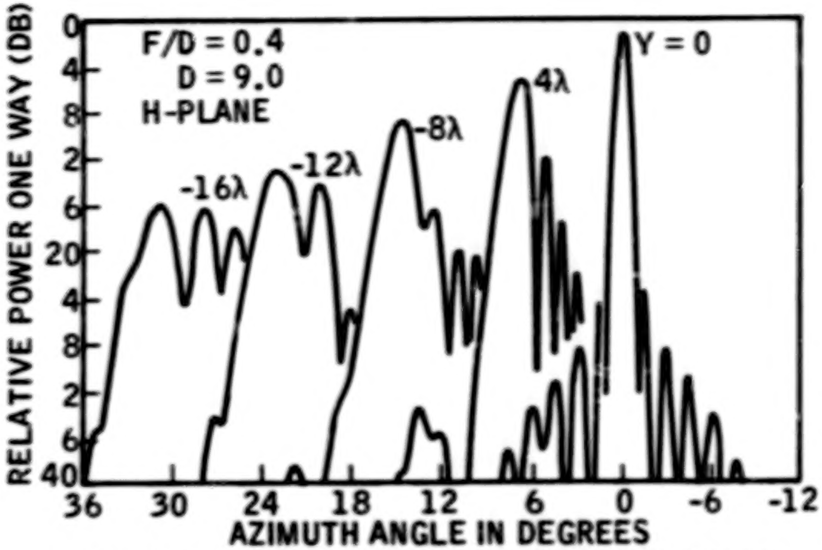
350

(Figure 8)

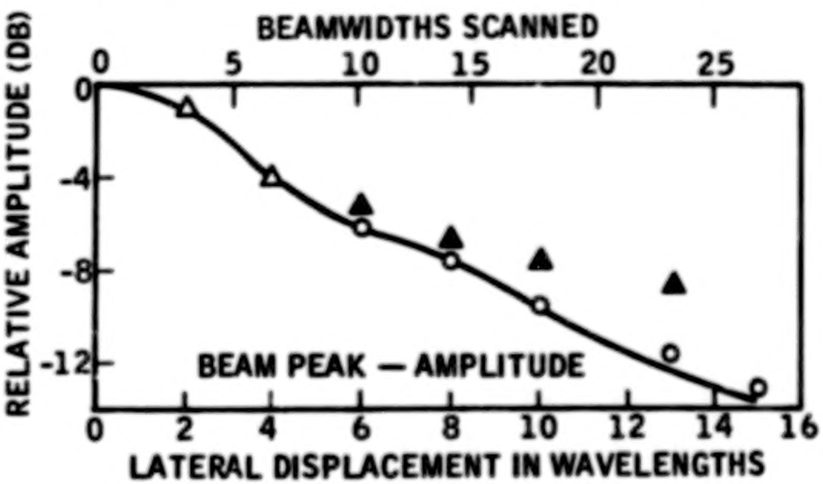
On a normal paraboloid, only one feed fits at the focal point. As other feeds are added for a single reflector, sidelobes increase and gain drops. As the sidelobe rises, interference into the next time zone will occur. With the excessive use of the frequency spectrum, regulatory agencies will prohibit the use of multiple beam systems with this problem.

This is the reason for selection of six reflectors for each time zone, and the use of only two feeds per reflector for the half-time zone coverage.

LIMITS ON NUMBER OF BEAMS ON HIGH POWER DIRECT TV BROADCAST SATELLITE ANTENNAS PER REFLECTOR

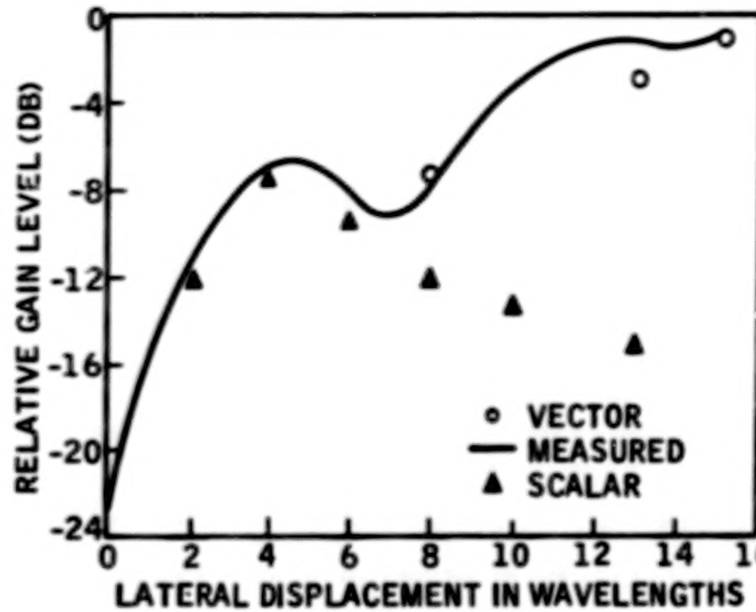


PATTERN DEGRADATION VS. LATERAL FEED DISPL.



GAIN LOSS VS. LATERAL FEED DISPLACEMENT

PATTERN DEGRADATION
POWER LOSS
INTERFERENCE



SIDELOBE INCREASE VS. LATERAL FEED DISPLACEMENT

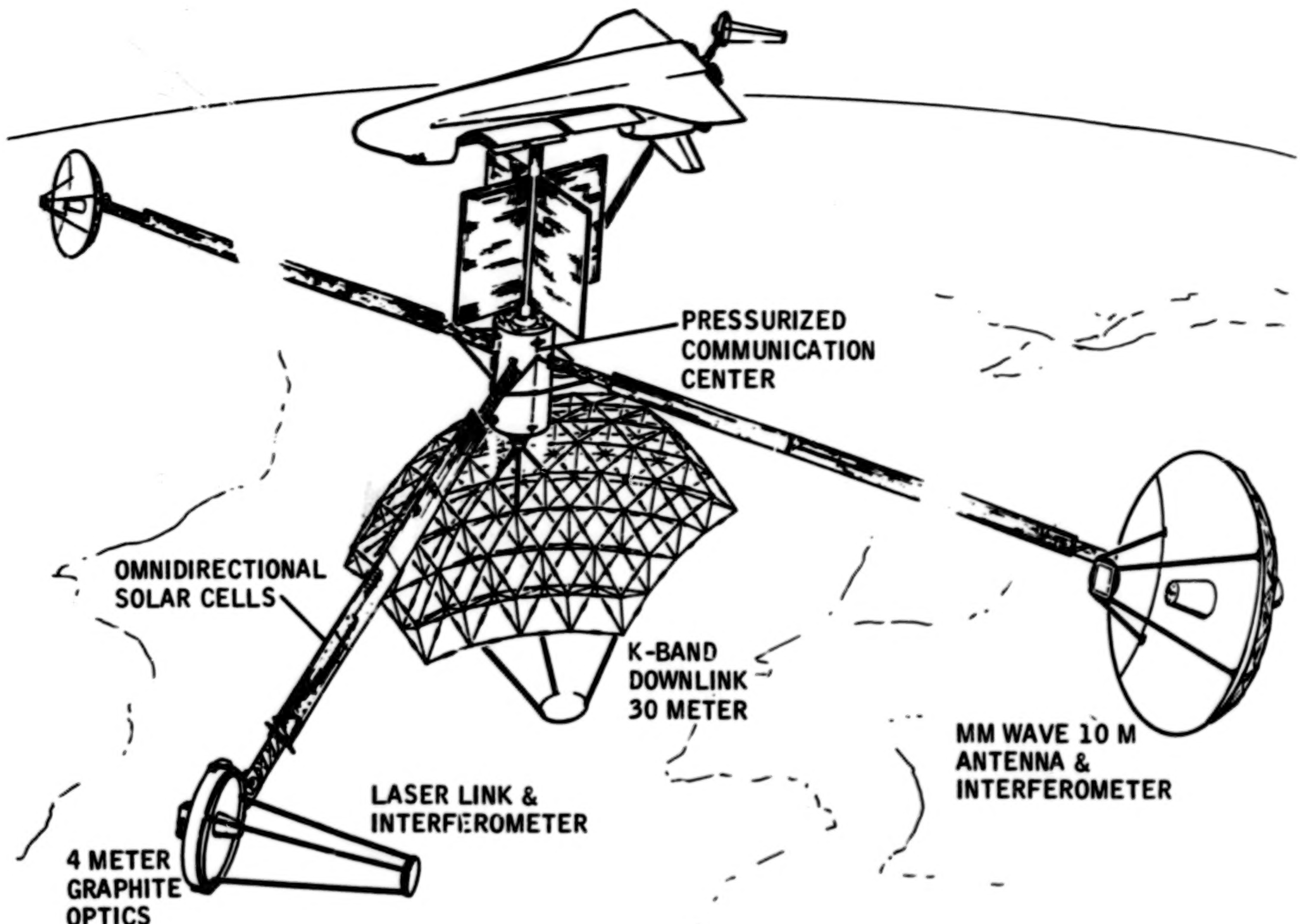
Figure 8

ORBITING DEEP SPACE COMMUNICATION CENTER (Figure 9)

In the 1985 - 2000 period, the deep space communication net could be controlled from an orbiting station. On one boom, two 10 meter diameter 60 GHz (minimum earth interference) would serve as independent communication links, or coupled perform interferometric measurements. The second boom would contain two 4 meter optical systems for laser communication systems in the link or interferometer mode. Processing and conversion to a large high-gain 30-meter K-band antenna would serve as the space-to-ground link. Omnidirectional solar cells would provide the power to the system with minimum perturbation on system pointing accuracy. All components, including the optics, would be fabricated from near-zero thermal expansion composite with high modulus in 40 kwi range to achieve stiffness necessary for pointing (0.04 deg for mm antenna and 0.0001 deg for laser system).

An advanced high orbit manned shuttle would maintain the electronics, lasers and transmitter. In operation the system would be unmanned. Orbit position could vary from 600 nmi, synchronous or lunar libration points depending on best position for the deep space probes of the period.

ORBITING DEEP SPACE COMMUNICATION CENTER



353

Figure 9

ORBITING DEEP SPACE COMMUNICATION CENTER

NEED

- PROVIDE COMMUNICATION & TRACKING FUNCTIONS FOR DEEP SPACE MISSIONS

CONCEPT

- 60 GHz ANTENNA (75 dB GAIN) FOR SPACE-TO-SPACE COMMUNICATION FOR NORMAL DATA RATES (LOW INTERFERENCE - O_2 ABSORPTION BAND)
- LASER FOR HIGH DATA RATE COMMUNICATIONS AT PLANETARY INTERCEPT
- INTERFEROMETRIC TRACKING CAPABILITY ON BOTH SYSTEMS
- CONVERSION TO K-BAND FOR DATA LINK
- PRESSURIZED COMMUNICATION CENTER ALLOWS DOCKING FOR MAINTENANCE (NOT MANNED IN OPERATION)
- SOLAR CELL PROTECTED PASSAGE TO LASER & ANTENNA

INNOVATIONS

- LOW EXPANSION GRAPHITE OPTICS
- 1-MIL RMS GRAPHITE MM ANTENNA
- ZERO EXPANSION, HIGH MODULUS BOOMS

Figure 10

356

(Figure 11)

An eight-foot diameter graphite composite mm wave antenna with 2.5 mil rms contour has been built by General Dynamics/Convair. Significant technology improvement is necessary to increase this hardware to an erectable concept with equivalent accuracy.

LARGE PRECISION GRAPHITE/ EPOXY SYSTEMS HAVE BEEN BUILT

C-BAND TO 300 GHz FREQUENCY COMMUNICATION ANTENNAS ARE FEASIBLE WITH GAIN CAPABILITY TO 70 dB

MECHANICAL PERFORMANCE

NATURAL FREQUENCY, Hz	55
INERTIA LOADING, g	30
SURFACE CONTOUR:	
MFG., IN.	0.0025
ORBITAL, IN.	0.0031
WEIGHT, LB	45
ACOUSTICS, dB	147
ABSORPTANCE/EMITTANCE	0.7

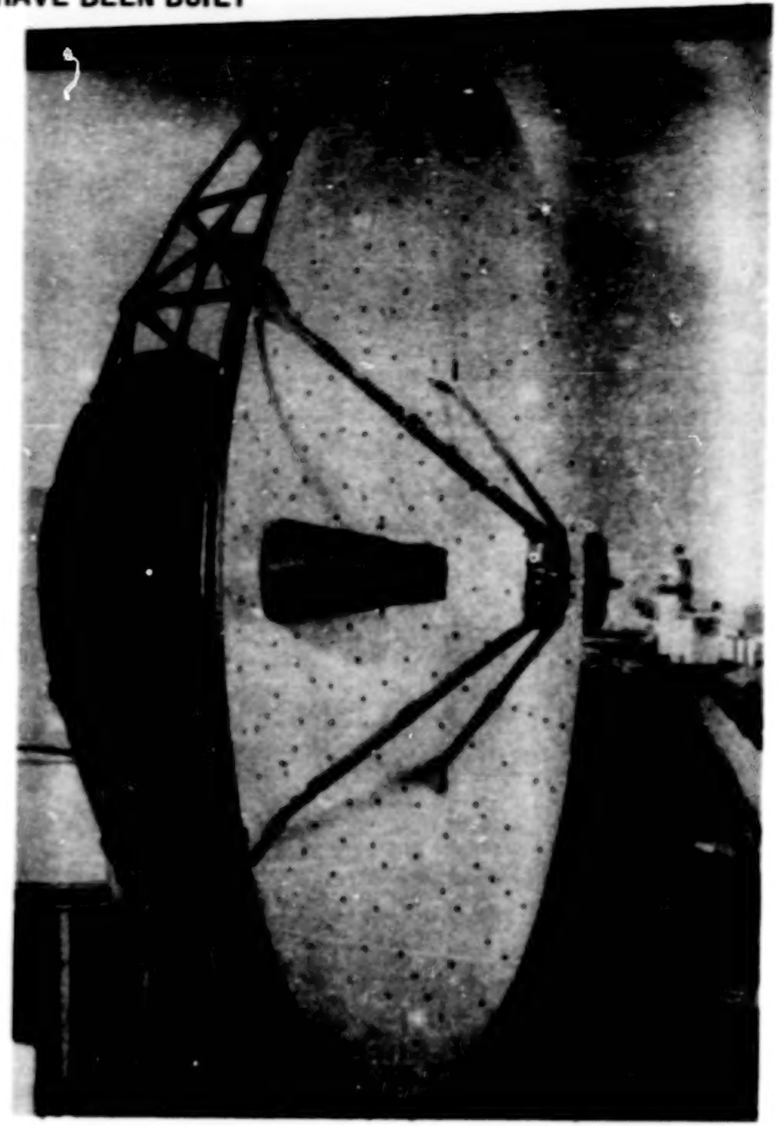


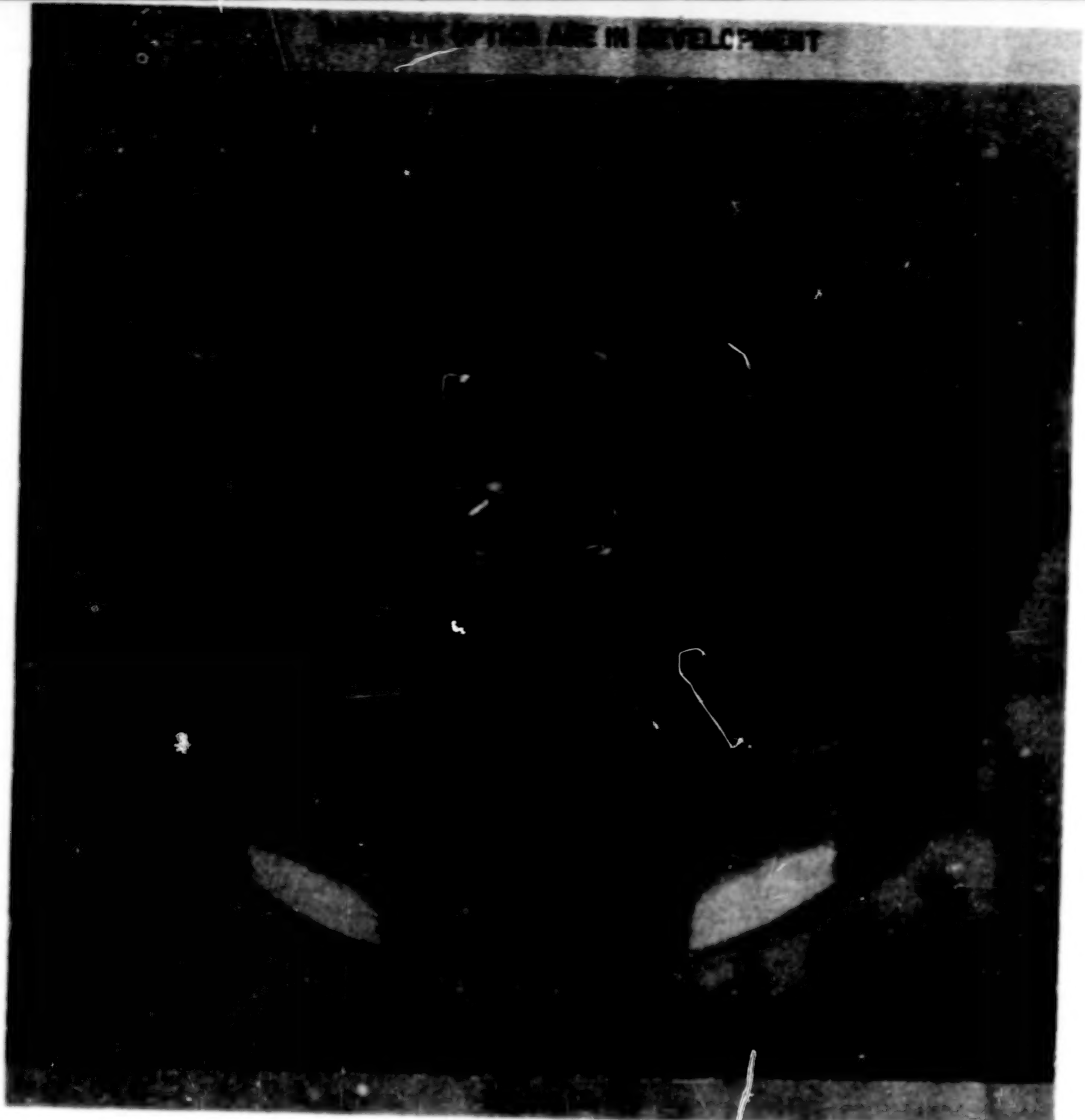
Figure 11

357

358

(Figure 12)

General Dynamics/Convair is developing lightweight graphite composite optics that could be used in large laser communication systems. Use of all composite faceplates would significantly reduce the mass moment of inertia of a large optics system.



359

COASTAL WATER SURVEILLANCE (Figure 13)

Radar satellites would orbit over the central U.S. at an altitude that would provide a 20 to 30 degree intercept angle to the coastal waters. Its radar would operate in the 1 - 3 GHz frequency range. The Atlantic and Pacific areas would be covered by separate synthetic aperture antennas that provide three high gain, narrow beams over the 200 mile limit. Gulf coverage would be provided by separate large paraboloids that would cover the Gulf horseshoe area each pass, then switch to Alaskan waters. It would also have the capability of retasking to cover "hot spots" in the Atlantic or Pacific areas.

With added power, it could also serve a similar function for Canada and Mexico.

COASTAL WATERS SURVEILLANCE RADAR SATELLITE

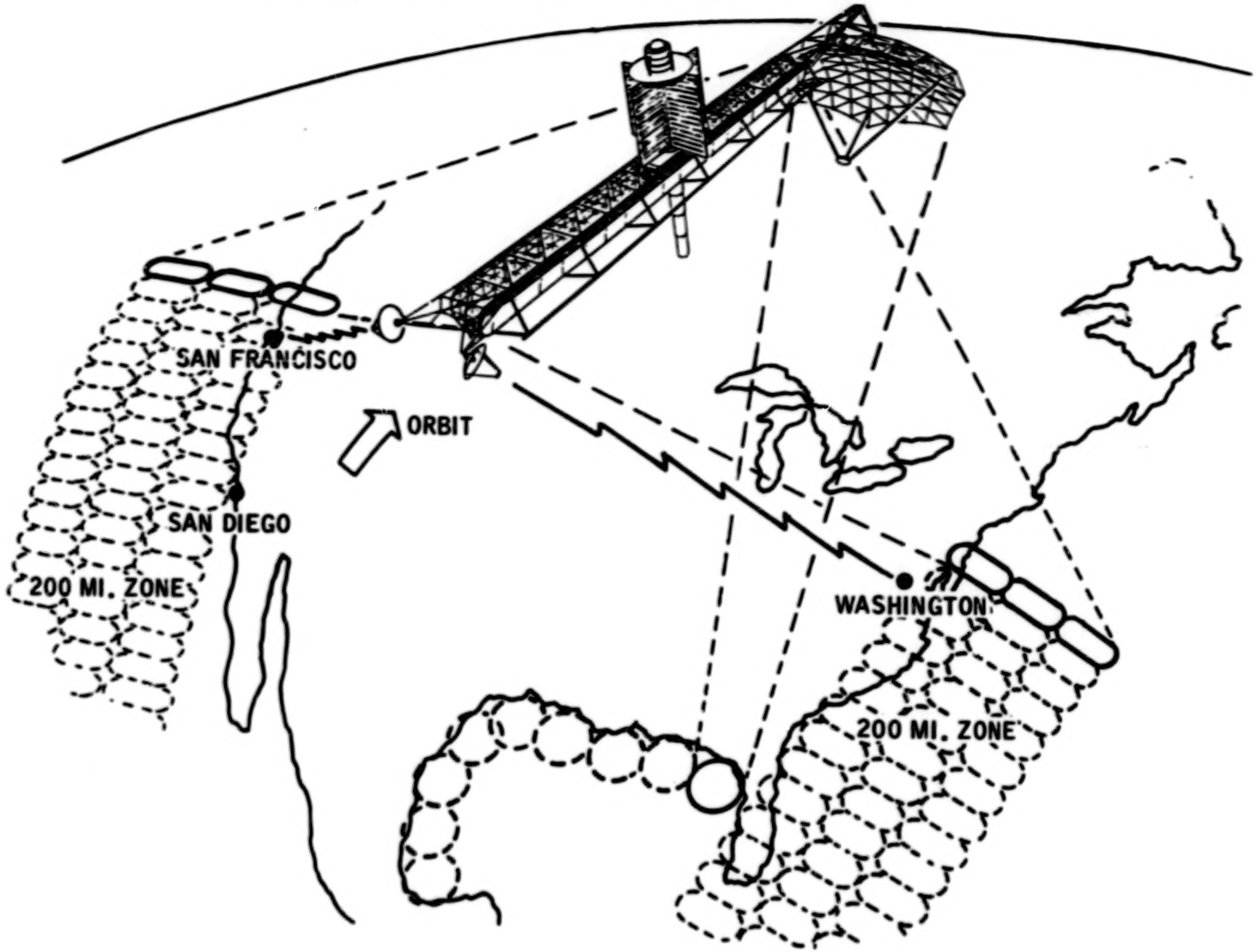


Figure 13

COASTAL WATER SURVEILLANCE SYSTEMS

NEED:

- DUE TO EXPANSION TO 200 MILE LIMIT, COST OF RADAR SATELLITE BECOMES COMPETITIVE COMPARED TO AIRCRAFT & SHIP PATROLS
- SYSTEM PROTECTS MARINE LIFE & MARINE FARM AREAS (FISHING VESSELS, ETC).
- TRAFFIC CONTROL – HIGH SPEED HYDROFOIL/AIR EFFECT VEHICLES MIXED WITH STANDARD VESSELS (TANKERS, ETC.) & OFF SHORE MINING RIGS.
- ECONOMIC CONDITIONS (TARIFFS) MAY MAKE SMUGGLING OF DEVELOPING NATION'S PRODUCTS A PROBLEM.

GENERAL REQUIREMENTS:

- 2 MILE INTERCEPT ACCURACY
- 1 HOUR REVISIT TIME
- .90 PROBABILITY OF INTERCEPT SMALL VESSELS (100 M^2)
- OPERATE UP TO 3 - 5 SEA STATE

Figure 14

COASTAL WATERS SURVEILLANCE RADAR SATELLITE

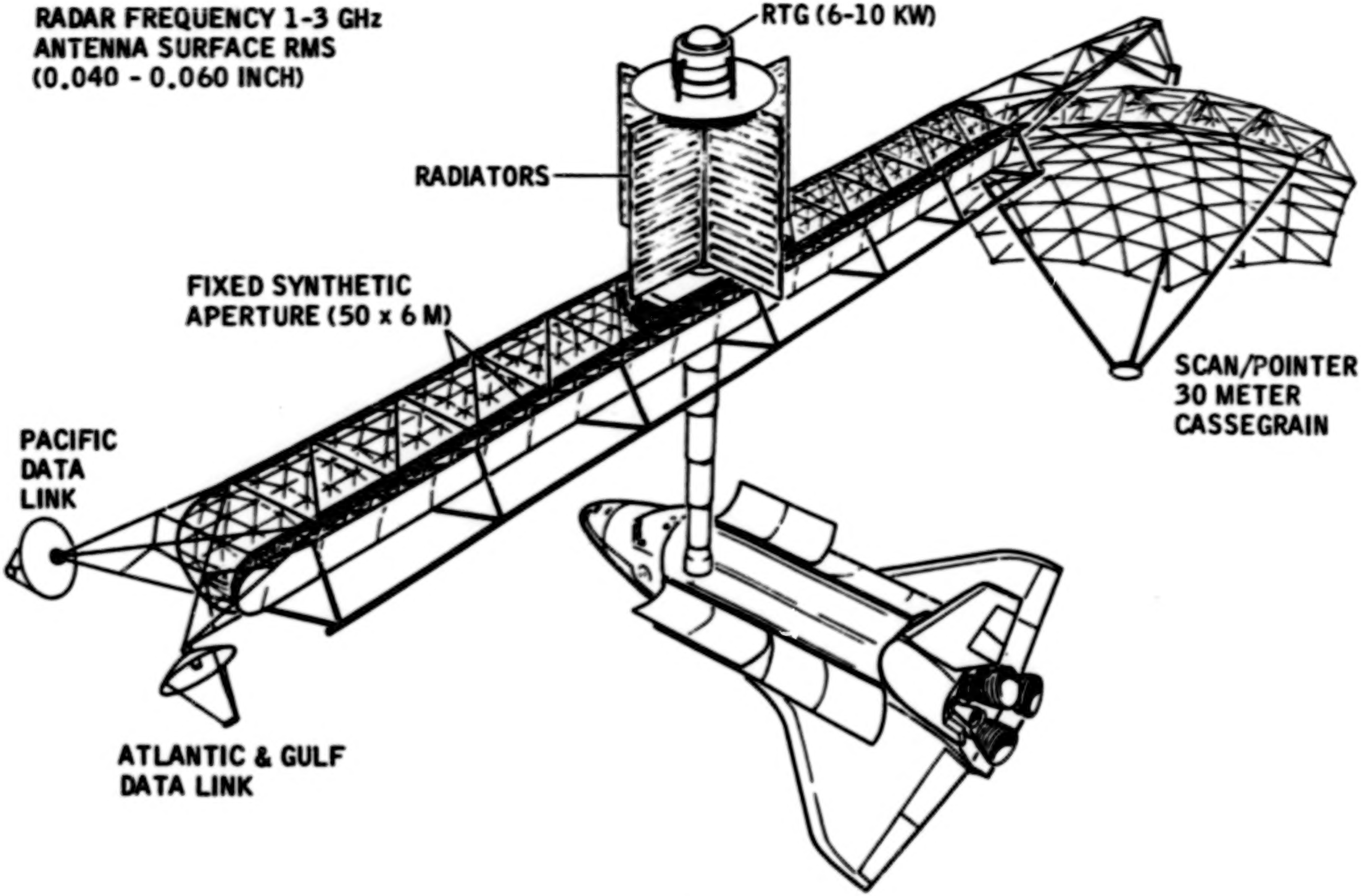
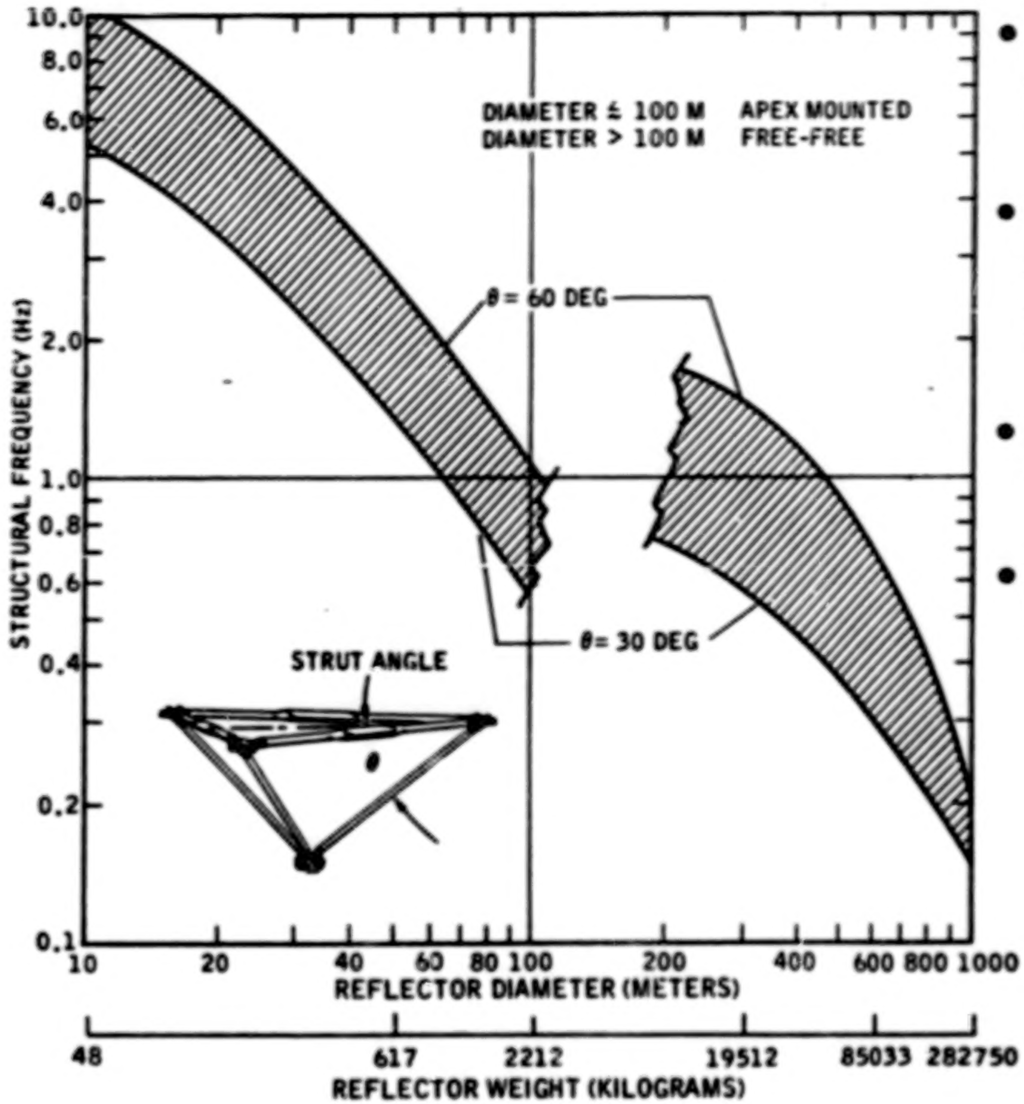


Figure 15

364

(Figure 16)

Geo-truss has the capability of providing the basic structural elements for the large antenna proposed in the three concepts. Stiffness is a major criteria to enable the structure to perform the rigorous pointing requirements of these narrow beamwidth systems. Stiffness greater than 2 Hz is needed if proven attitude control system concepts are to be used. At low frequency, complex distributed control systems may be needed to reach our goals.



- LARGE ANTENNAS MEAN NARROW BEAMWIDTH & RIGOROUS POINTING REQUIREMENTS
- INTERNATIONAL AGREEMENTS (CCIR) REQUIRES BEAMS & SIDELOBES BE MAINTAINED WITHIN ASSIGNED BOUNDARY
- REASONABLE STIFFNESS CAN BE OBTAINED WITH GEO-TRUSS ANTENNA
- OVER 100 M ANTENNA & SPACECRAFT BECOMES FREE-FREE STRUCTURE

Figure 16

365

(Figure 17)

General Dynamics/Convair has produced erectable geo-truss concepts in multiple combinations of the basic tetrahedron building block. All have been successfully deployed and rf tested. Mesh contour of 15-mil rms has been achieved which allows the concept to efficiently operate in the 20 - 30 GHz regions.

DEPLOYMENT OF MULTIPLE ELEMENT GEO-TRUSS HAS BEEN REPEATEDLY PROVEN

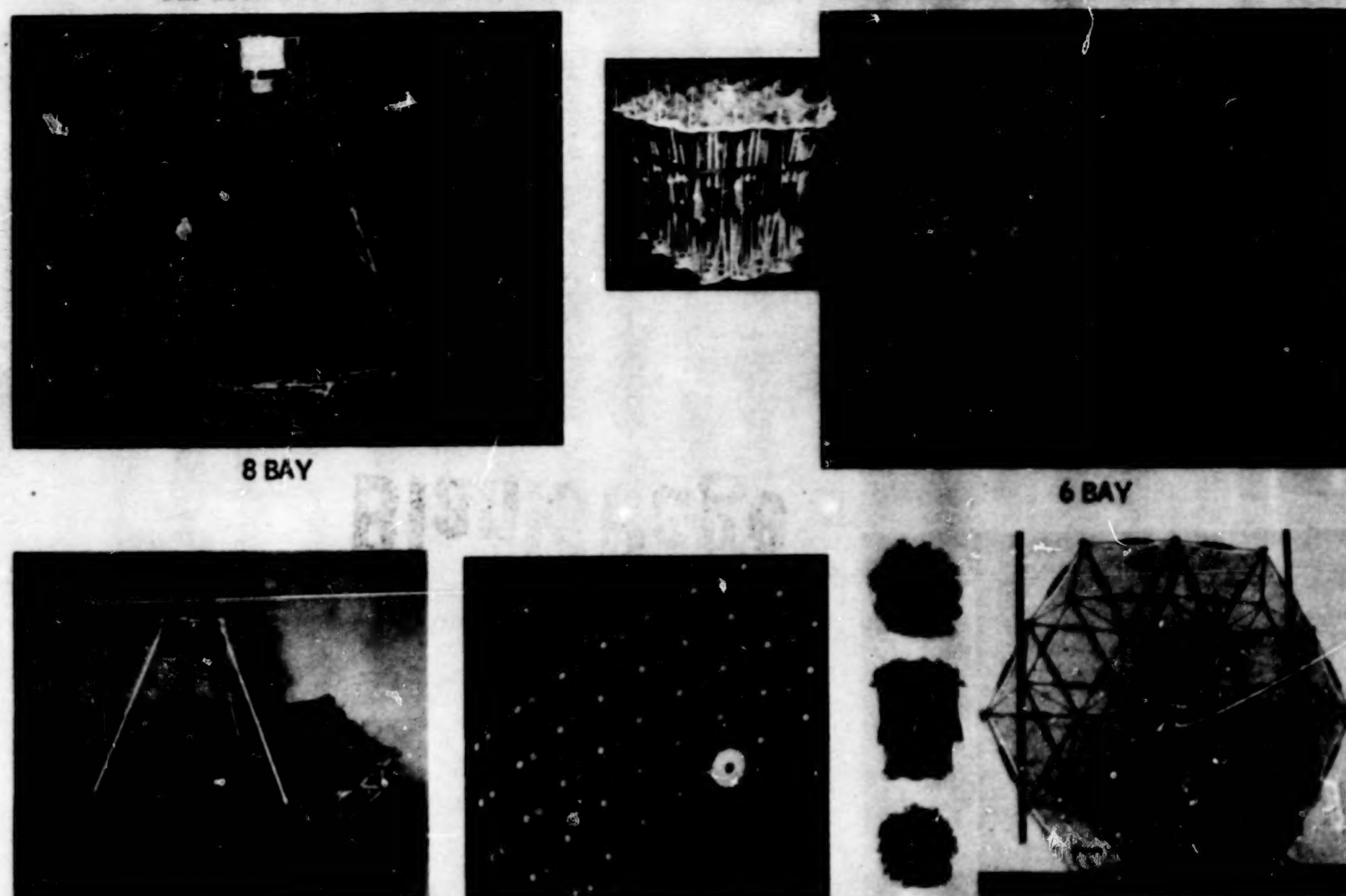


Figure 17

368

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USAF ANTENNA
ON-ORBIT ASSEMBLY

PRESENTED AT

LANGLEY RESEARCH CENTER

TO THE

LARGE SPACE SYSTEMS TECHNOLOGY SEMINAR

BY

CAPTAIN PAUL E. HEARTQUIST

SPACE AND MISSILE SYSTEM ORGANIZATION

369

370

SAMSO ON-ORBIT ASSEMBLY CONTRACTS (Figure 1)

On-Orbit Assembly presentation was given by Captain Paul E. Heartquist, On-Orbit Assembly Project Manager, Space Transportation System Program Office, Space and Missile System Organization (SAMSO), Los Angeles, California 90009.

"Phase-One" will be defined on a later page.

Two Phase-One (Concept Design) contracts were initiated on 17 Oct 78 by the General Dynamics Corporation-Convair Division and by the Martin Marietta Corporation.

The first program milestones were the Concept Reviews on 5 and 12 Jan 78.

Spacecraft designs applicable to the 300-1000 feet range will be extrapolated from the design of a 600-foot spacecraft. Subsequent phase of program will involve the preliminary design of a Flight Demonstration Article (FDA).

Flight Demonstration of automated assembly will be accomplished with manned oversight in the vicinity of the Space Shuttle orbiter. This will allow for spacecraft repair/recovery, if required, as well as extravehicular activity (EVA) as a contingent repair mode.

The Flight Demonstration provides simulation of certain technical aspects to include the following:

- payload packaging
- shuttle effects on payloads
- deployment from shuttle
- module deployment
- subsystem deployment
- rendezvous
- module control
- docking
- surface alignment
- spacecraft control
- sensor performance

SAMSO ON-ORBIT ASSEMBLY CONTRACTS

- PARALLEL PHASE-ONE DESIGN CONTRACTS
  - / FIRM FIXED PRICE
  - / $750K PER CONTRACT
  - / CONTRACTORS ARE GDC & MMC
- PROVIDE STRUCTURAL/SPACECRAFT DESIGN APPLICABLE TO 300-1000 FEET
  - / DESIGN 600 FOOT SPACECRAFT
  - / PRELIMINARY DESIGN FDA
- DEMONSTRATION IN ORBIT
  - / FDA AUTOMATED ASSEMBLY
  - / STS CAPABILITY

Figure 1

372

ON-ORBIT ASSEMBLY PROGRAM PLAN (Figure 2)

This page depicts the general program layout which allows flexibility should modification be desired.

The program may also include other USAF technology demonstrations (i.e., materials, electronic modules and antenna design) if funds are made available.

Phase III equates to a Full-Scale Development phase.

ON-ORBIT ASSEMBLY PROGRAM PLAN

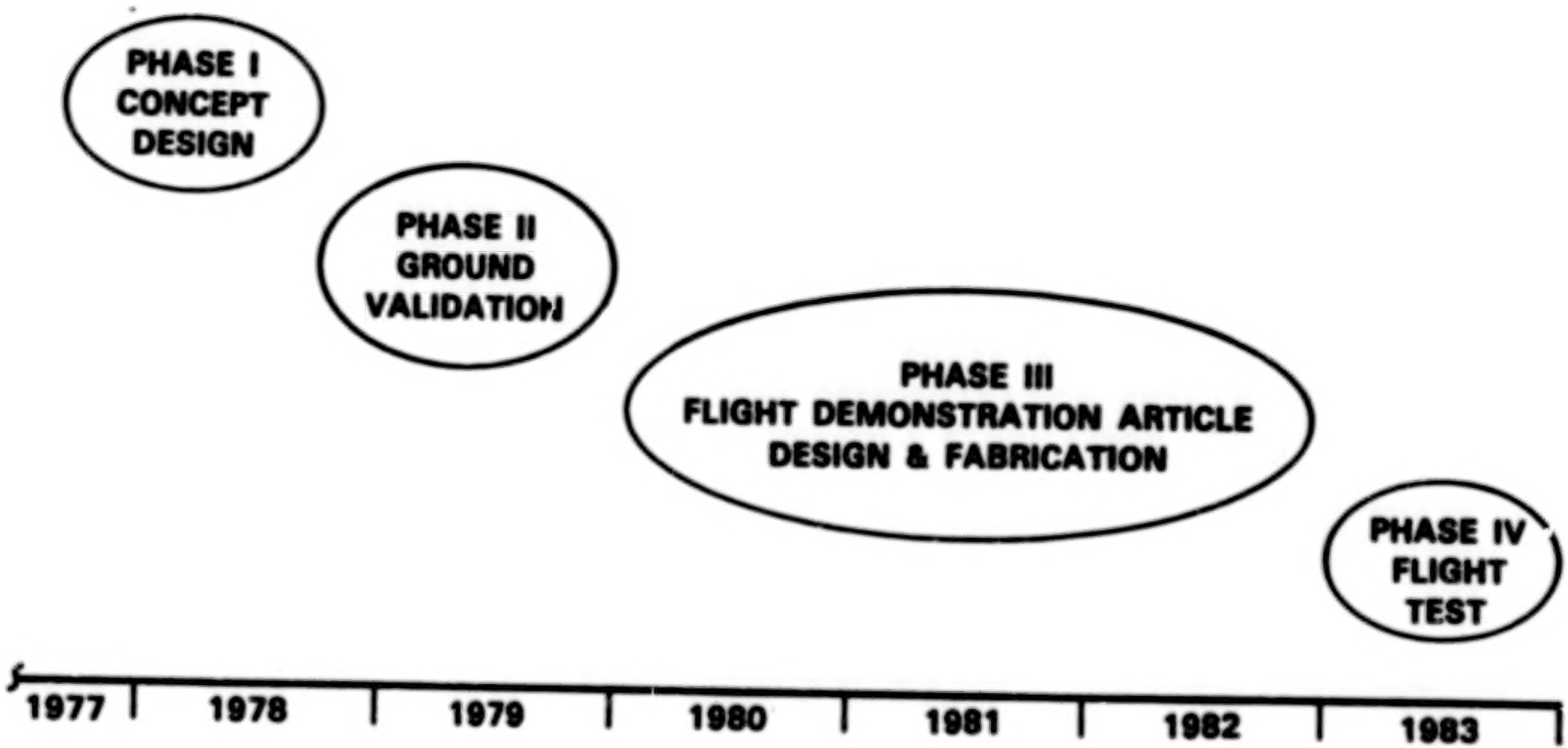


Figure 2

373

CONTENTS

VOLUME I

PROGRAM OBJECTIVES (Figure 3)

The overall On-Orbit Assembly program objectives and ground rules are outlined on this page.

The structure is deployed and checked out in the near vicinity of the Shuttle to facilitate repair, if required. The requirement for repair is quite likely considering the very large number of deploying parts. This repair could be automated or, as a contingency, through the use of EVA.

The subject of upper stages will be addressed in greater detail later.

Examples of other large spacecraft requirements (other than radar) would be communications or large optics.

Program Objectives and Requirements

Develop Shuttle-Based, On-Orbit Assembly Technique Applicable to Spacecraft with 300- to 1000-ft-Diameter Sensors

Deploy and Check Out in Near Vicinity to Shuttle

Transport Deployed Array or Array Sections to High Earth Orbit (HEO) Using One or More of the Specified Upper Stages

Rendezvous/Dock/Activate in HEO

Use Minimum Shuttle Launches

Adaptable to Other Large Spacecraft Requirements

Figure 3

376

CONCEPT DESIGN STUDY (Figure 4)

The Phase One, Concept Design Study, is the current effort of the contractors and should be discussed in greater detail.

The efforts that have been completed are those involved with the design concept selection (i.e., structural concepts, upper stage evaluations, and orbiter packaging considerations.)

The contractors are now ready to begin the preliminary spacecraft design and risk analysis tasks.

Once these efforts are complete, the preliminary design considerations of the Flight Demonstration System will be addressed and the contractors will establish ground validation equipment requirements and develop a Phase II (Ground Validation) Plan.

CONCEPT REVIEW

PHASE I
CONCEPT DESIGN STUDY

- CONCEPT SELECTION
  - — SPACECRAFT CONCEPTS
  - — UPPER STAGE EVALUATIONS
  - — ORBITER PACKAGING
- PRELIMINARY SPACECRAFT DESIGN
- RISK ANALYSIS
- FLIGHT DEMONSTRATION SYSTEM
- GROUND VALIDATION EQUIPMENT
- PHASE II PLAN

Figure 4

DESIGN CONCEPTS (Figure 5)

The next few pages describe the design concepts presented to SAMSO.

Pictured on this page is the Martin Marietta concept shown in its stowed configuration in the Shuttle Orbiter bay. The stowed electronic array is shown immediately behind the aft flight deck, the upper stage (in this case a low thrust liquid) is in the aft end of the cargo bay and the stowed structure is between the array and upper stage.

The main consideration is packaging optimization using the full bay in order to minimize the required number of Shuttle flights.

Stowed Antenna with LTL Upper Stage

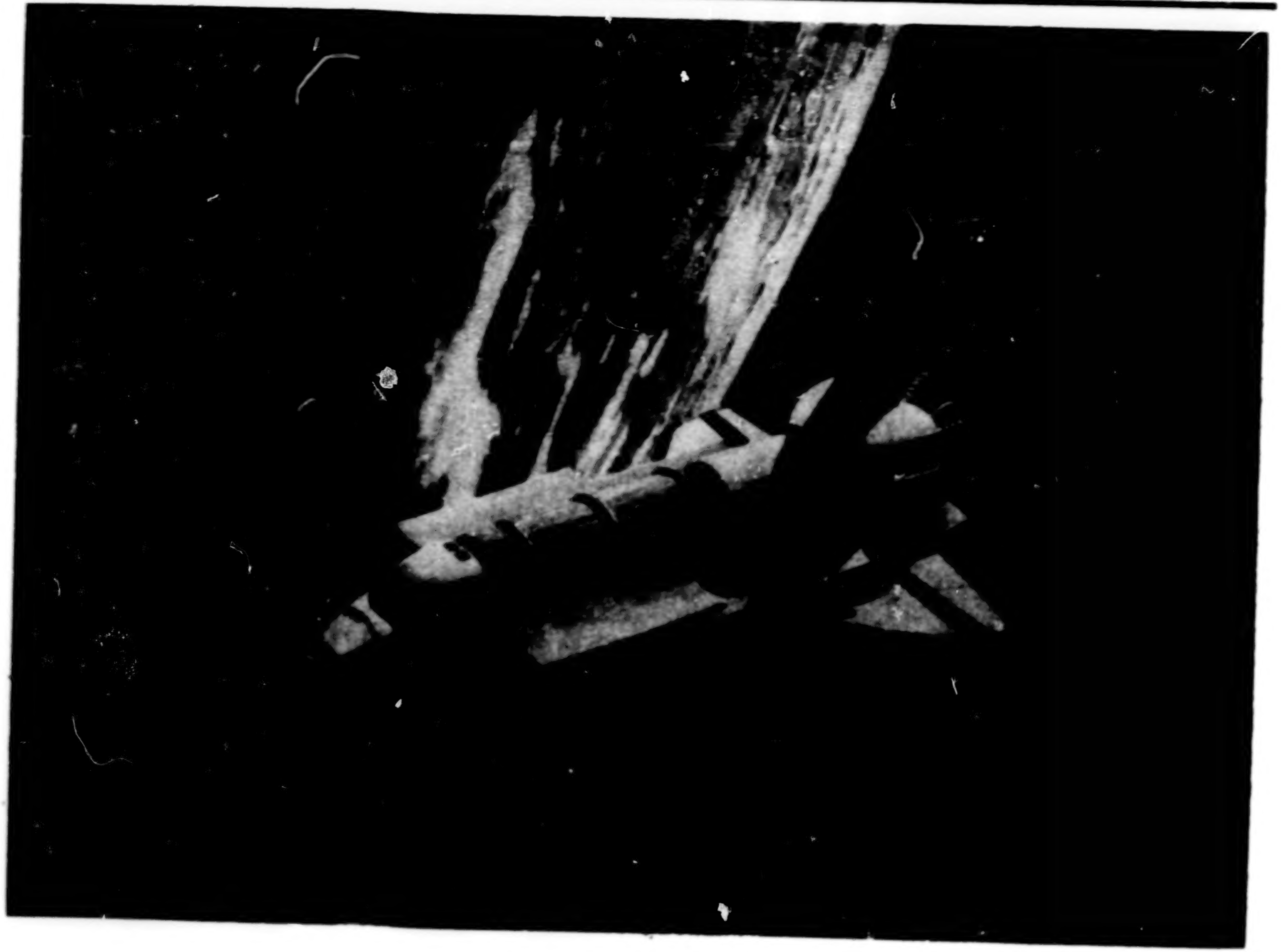


Figure 5

379

ARRAY DEPLOYMENT AND TRANSFER (Figure 6)

This page illustrates the Martin Marietta deployment scheme.

The stowed configuration (array, structure, and upper stage) are rotated out of the bay and deployed from the Shuttle.

Then, in the near vicinity of the Shuttle, the structure **expands** (first the columns deploy, then the rows).

Once deployed and checked out, the structure is transferred to high earth orbit (HEO) by the attached upper stage.

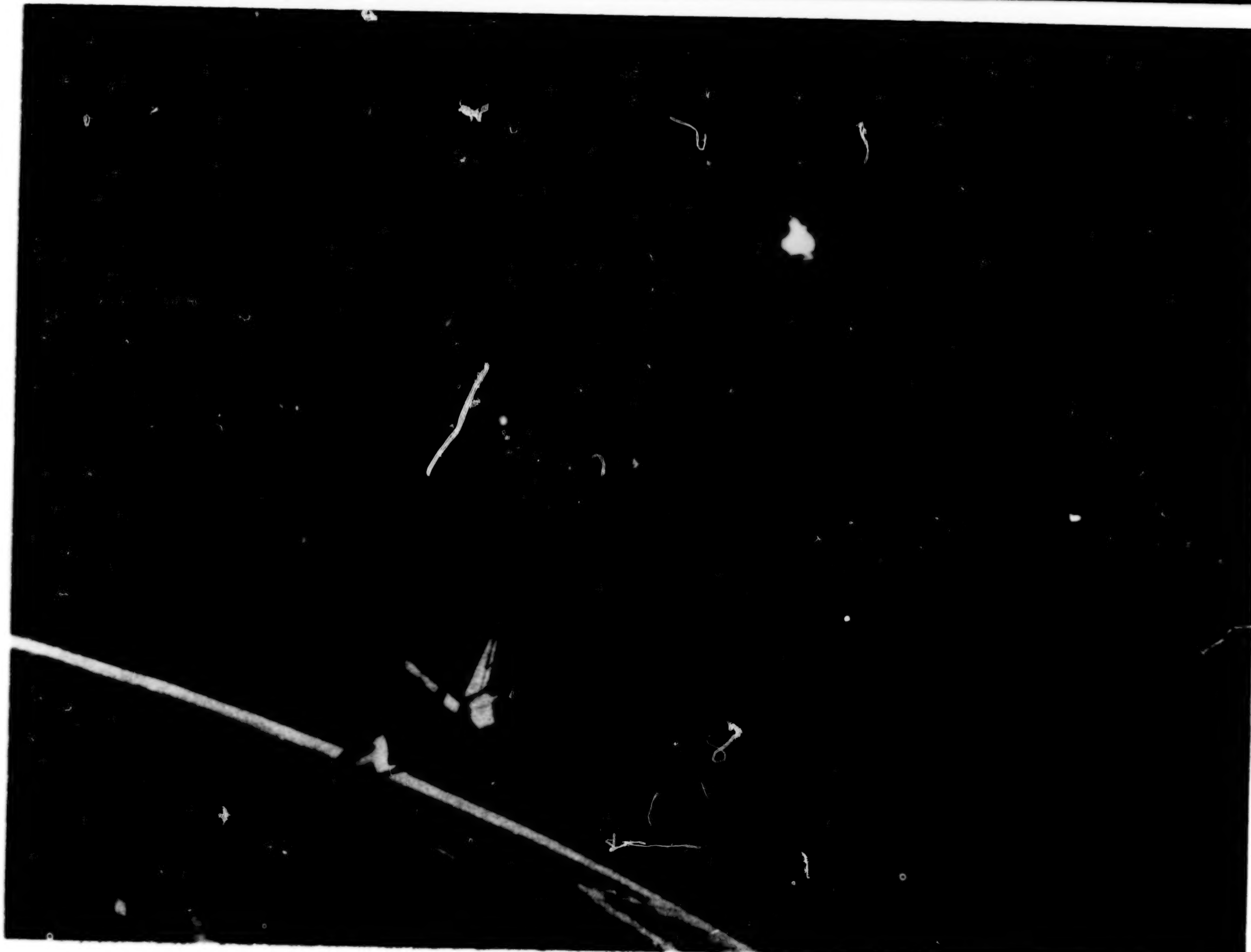


Figure 6

HEO RENDEZVOUS AND DOCKING (**Figure 7**)

In the cases where the required structure is sufficiently large to warrant more then one Shuttle flight, rendezvous/docking can be accomplished at HEO.

HEO Rendezvous/Docking

Figure 7

383

EXPANDABLE HEX CONCEPT (Figure 8)

This is one of three structural concepts presented by General Dynamics Corporation for SAMSO evaluation.

The deployment details of this concept are presented on the next page.

EXPANDABLE HEX CONCEPT

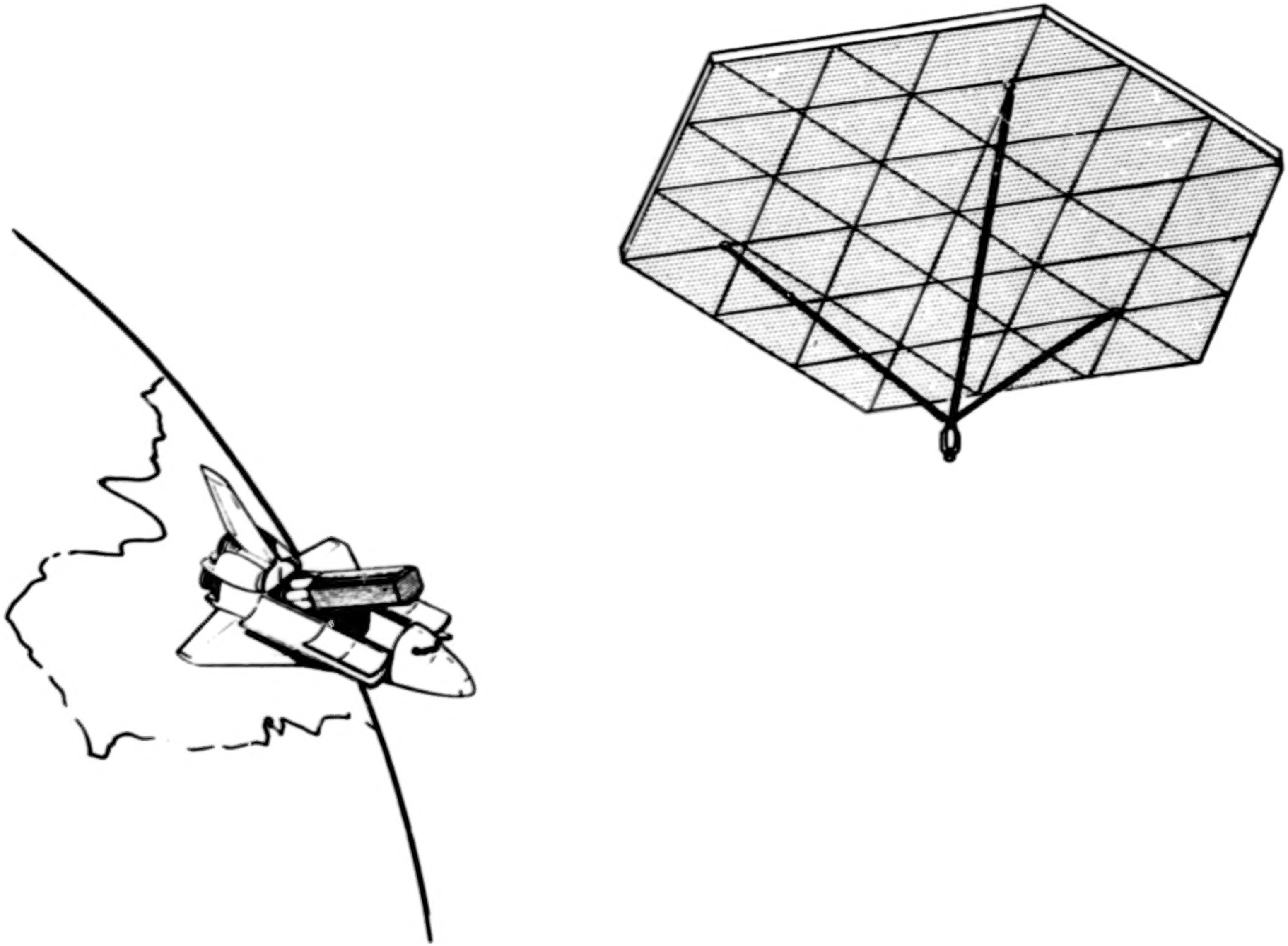


Figure 8

385

386

DEPLOYMENT DETAILS (Figure 9)

As shown, the stowed configuration consists of a stack of sections connnected by hinges.

The sections deploy laterally and become rigid to form a column.

The sections then deploy perpendicular to the "column" (shown by arrows) to form the large structural array.

CONCEPT A — DEPLOYMENT DETAILS

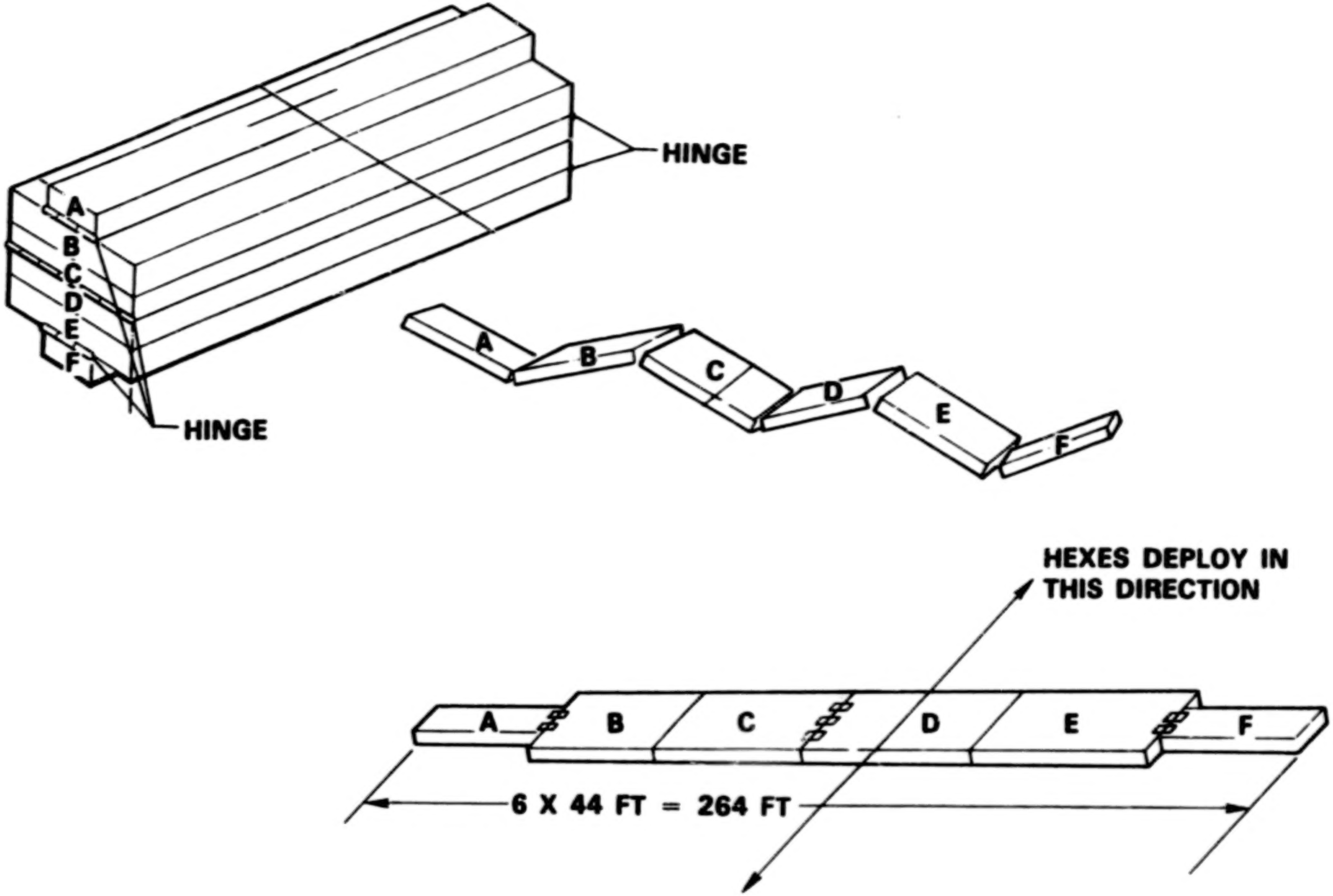


Figure 9

387

EXPANDING TETRAHEDRAL RING (Figure 10)

The second General Dynamic concept requires the structural array and the upper stage to be mounted on two separate cradles.

As shown, the stowed structure and upper stage are rotated so that they are perpendicular. They are then connected and deployed.

Note that the upper stage is connected to the edge of the section applying a thrust vector in the plane of the electronic mesh. This configuration protects the mesh from orbital transfer acceleration loads.

CONCEPT C —
EXPANDING TETRAHEDRAL RING

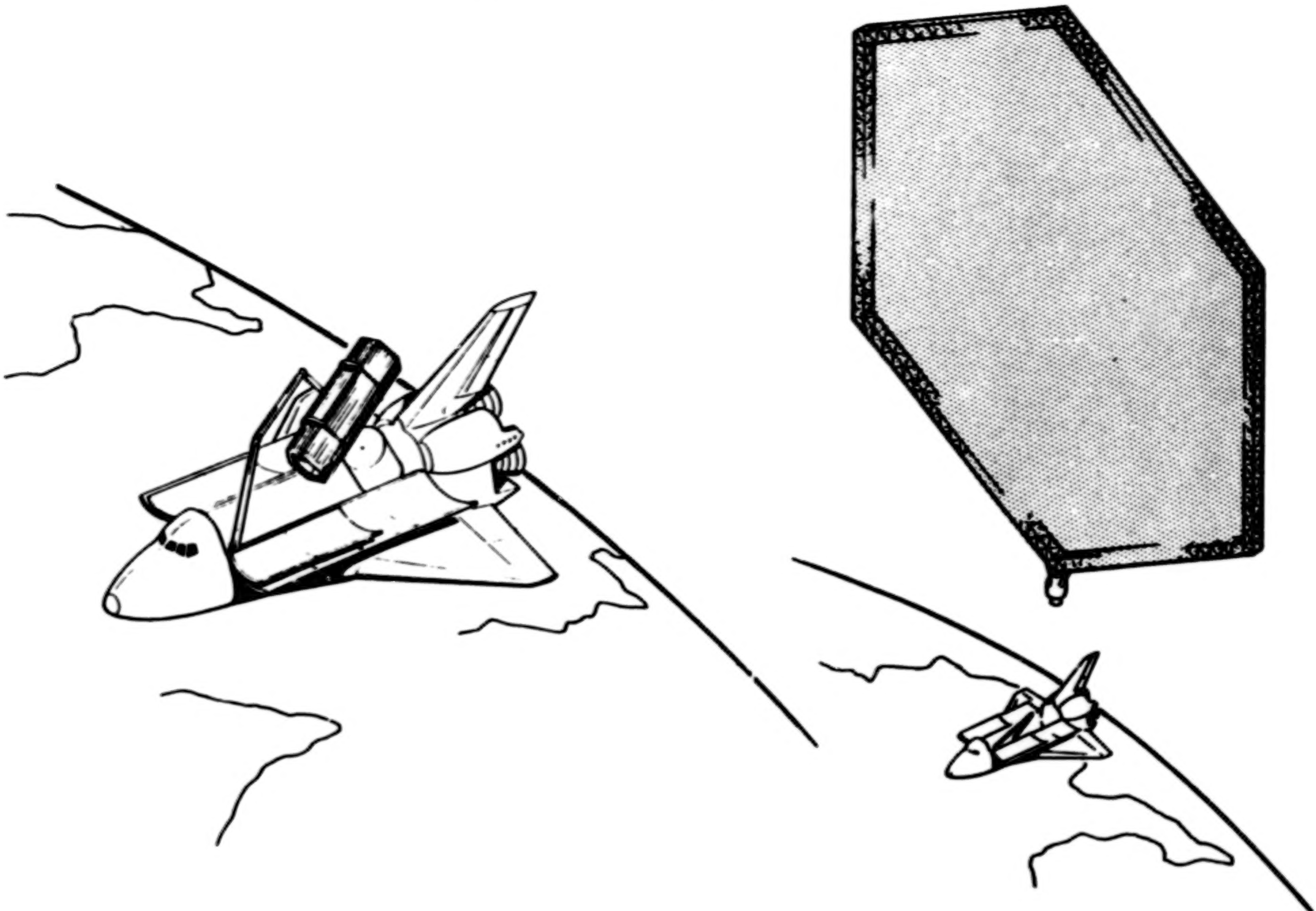


Figure 10

390

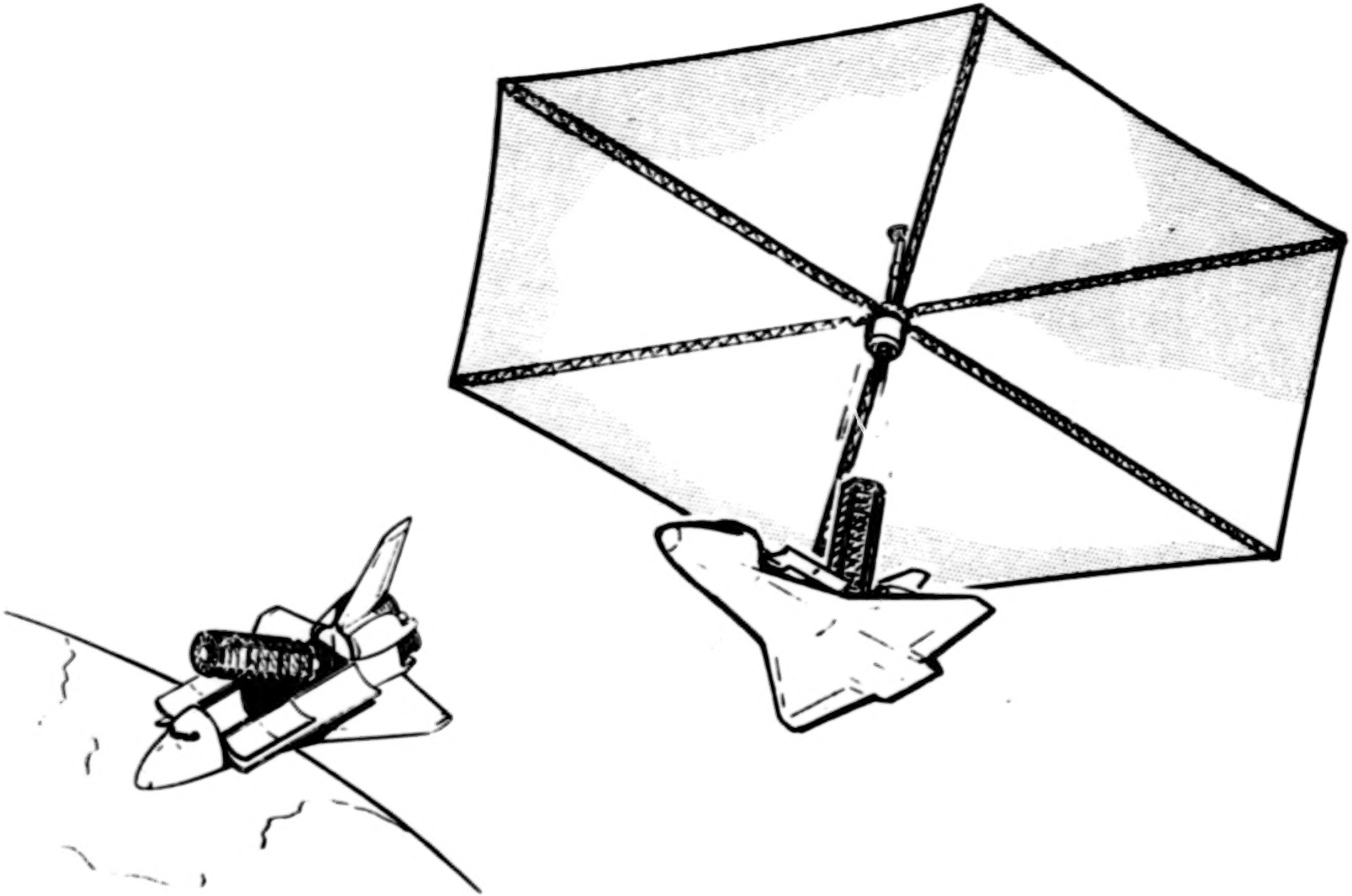
FOLD OUT TRUSS **(Figure 11)**

The third General Dynamic concept (and the concept that they recommend) makes use of a cradle that totally encloses the structure and upper stage while in the Shuttle Orbiter bay.

After the cradle rotates to a position perpendicular to the longitudinal axis of the Shuttle, the payload is elevated up through the cradle. The beams then fold down and expand - deploying the electronic mesh.

Once the spacecraft is disconnected from the Shuttle, and a safe separation distance is reached, the upper stage is ignited for orbital transfer.

CONCEPT E — FOLD OUT TRUSS



391

Figure 11

UPPER STAGE CONFIGURATIONS (Figure 12)

In addition to considering the design concepts described, SAMSO must select the upper stage most appropriate for this application.

The three types of upper stages being considered are Interim Upper Stages (IUS), low thrust liquids (LTL) and Solar Electric Power Stages (SEPS).

The thrust-to-weight ratios considered range from 0.0001g (SEPS) to 2.9g (IUS), with the LTL ranging from 0.02-0.4g.

Although only the cryogenic LTL is pictured here, the pressure-fed and the pump-fed storable LTLs were also considered (described on the next page).

Upper Stage Final Candidates—Configurations

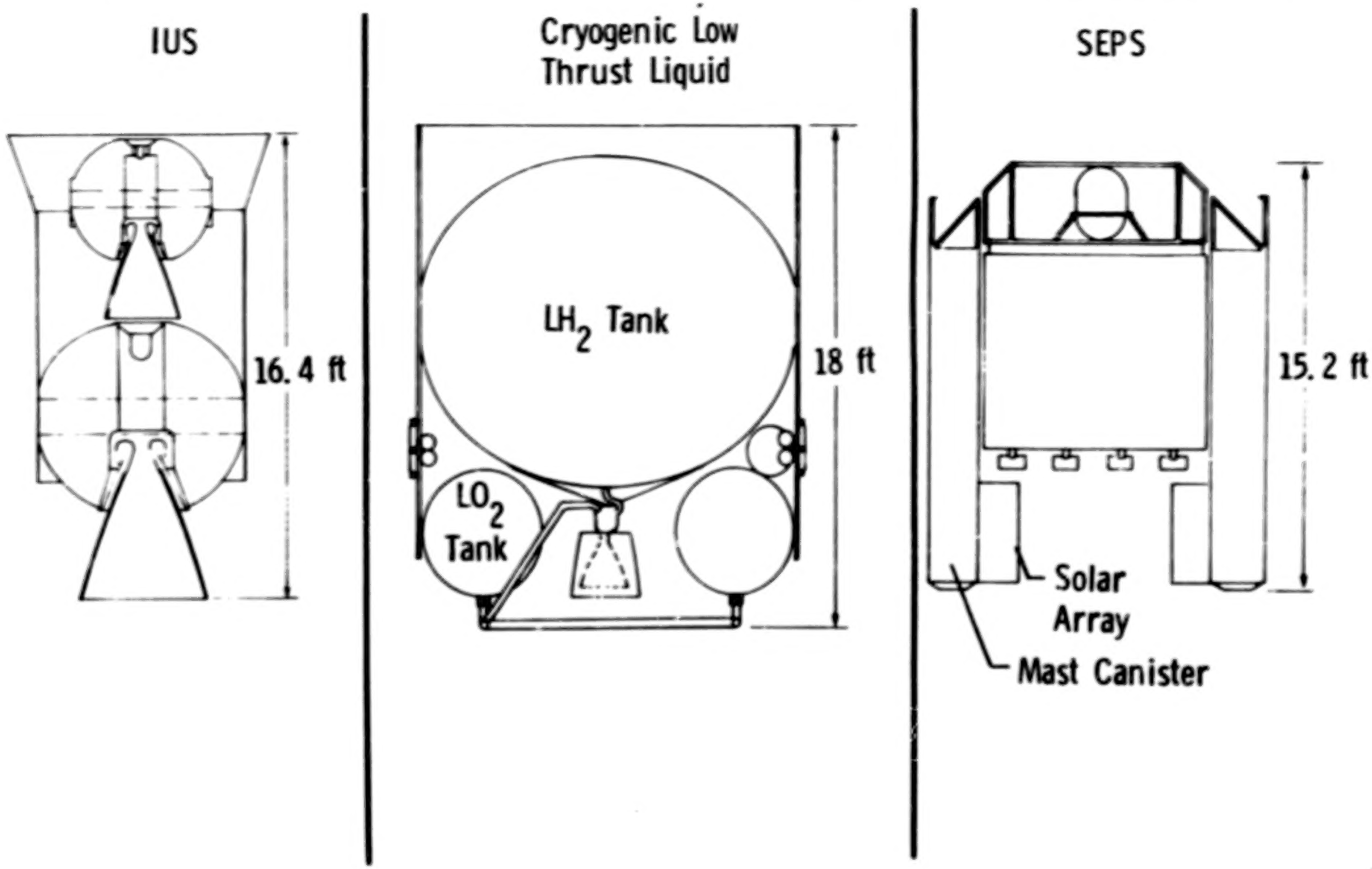
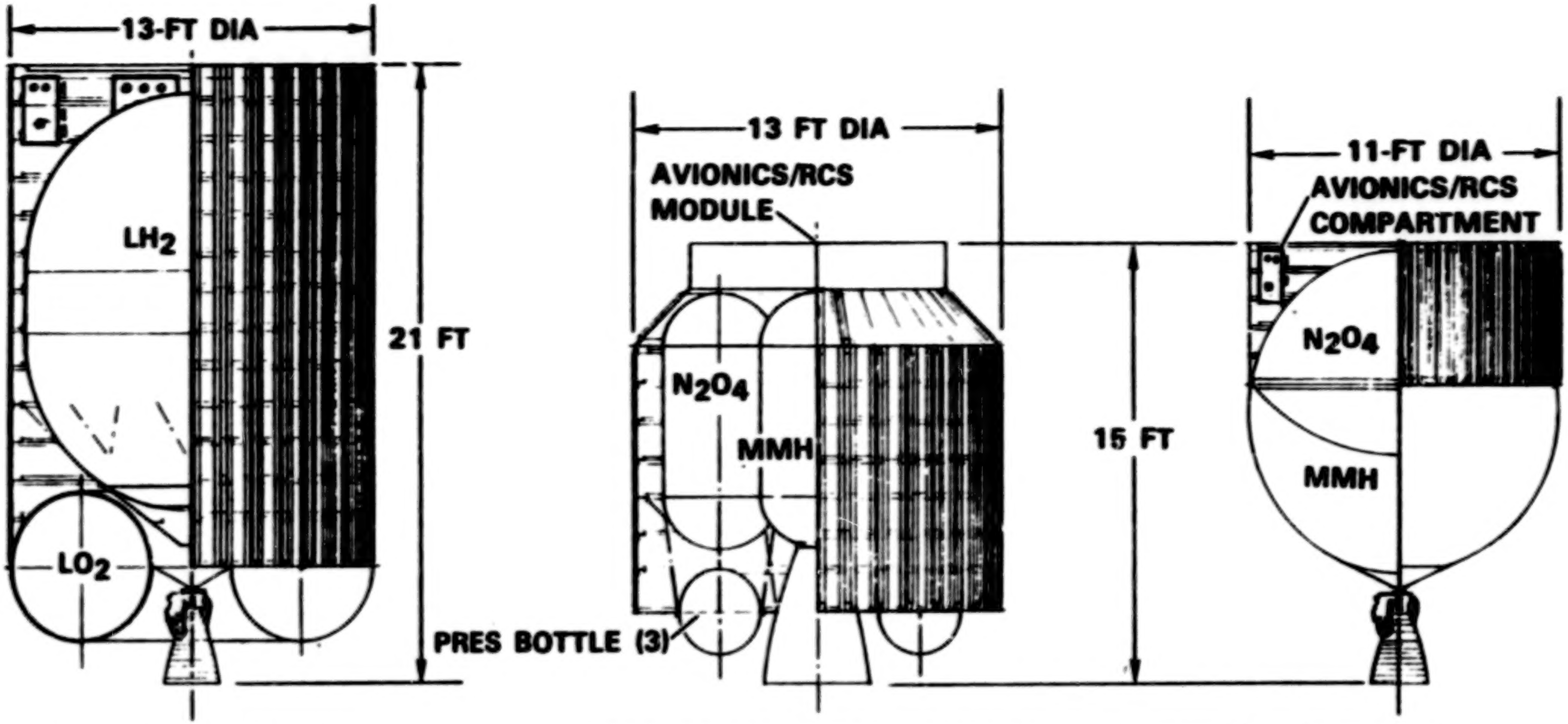


Figure 12

LOW THRUST LIQUID CONFIGURATIONS (Figure 13)

The most important data to observe while comparing the candidate upper stages are the T/W ratios and the associated payload capabilities.

LOW THRUST LIQUID STAGE CONFIGURATIONS



PUMP FED CRYOGENIC
40,000 LB LO$_2$/LH$_2$
M.F.= 0.85, ISP= 445
T= 1,000 LB$_f$, T/W= 0.05 g max
BURN TIME= 5 HR
PAYLOAD= 14,000 LB (4 BURN)

PRESSURE FED STORABLE (OMS)
46,000 LB N$_2$O$_4$/MMH
MF=0.85, ISP=310
T= 6,000 LB$_f$, T/W=0.4g max
BURN TIME=0.7 HR
PAYLOAD= 6,000 LB

PUMP FED STORABLE
47,000 LB N$_2$O$_4$/MMH
MF= 0.92, ISP= 320
T= 700 LB$_f$, T/W= 0.05 g MAX
BURN TIME= 6 HR
PAYLOAD= 9,000 LB (4 BURN)

Figure 13

UPPER STAGE INFLUENCE ON CONCEPT SELECTION* **(Figure 14)**

This graph shows the comparative T/W ratios of the candidate upper stages plotted against their payload capabilities to **geosynchronous altitude**.

The time values in parentheses below the various T/W values are estimates of the associated transfer times from low earth orbit to GEO.

The dashed lines (associated with the low-thrust-liquid stages) show the advantage of multiple perigee burns.

*The "Upper Stage Influence on Concept Selection" graph on this page was provided by General Dynamics Corporation - Convair Division.

UPPER STAGE INFLUENCE ON CONCEPT SELECTION

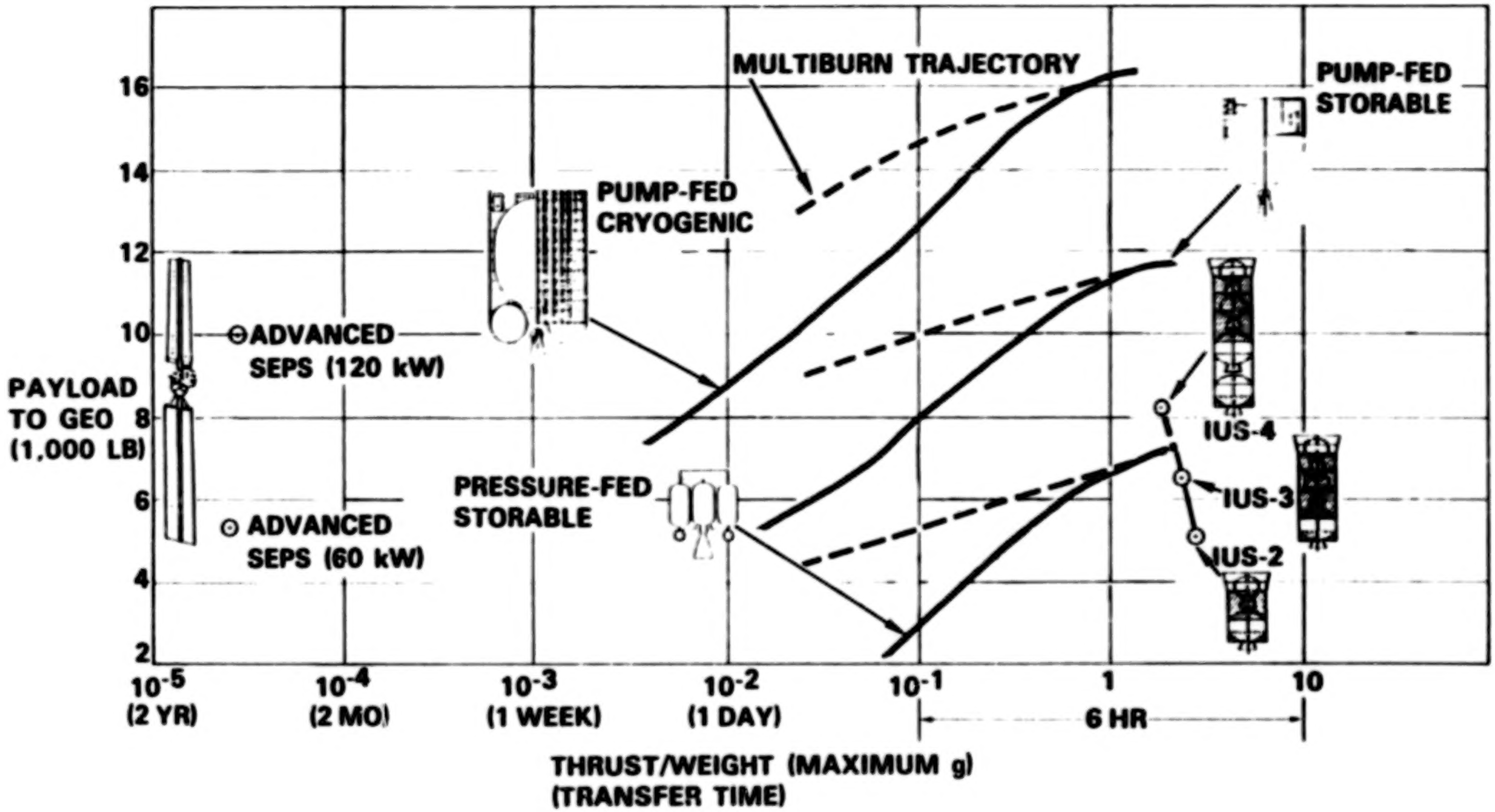


Figure 14

397

MAXIMUM DIAMETER ARRAY* (Figure 15)

The graph shown is similar to the one on the previous page except that the vertical axis shows the maximum diameter array that the stage can transfer from low to high earth orbit. (SEPS case does not include the weight penalty of any radiation shielding that may be required)

*The "Maximum Diameter Array" graph was provided by Martin Marietta Aerospace, Denver Division.

Maximum Diameter Array vs Thrust/Weight Ratio

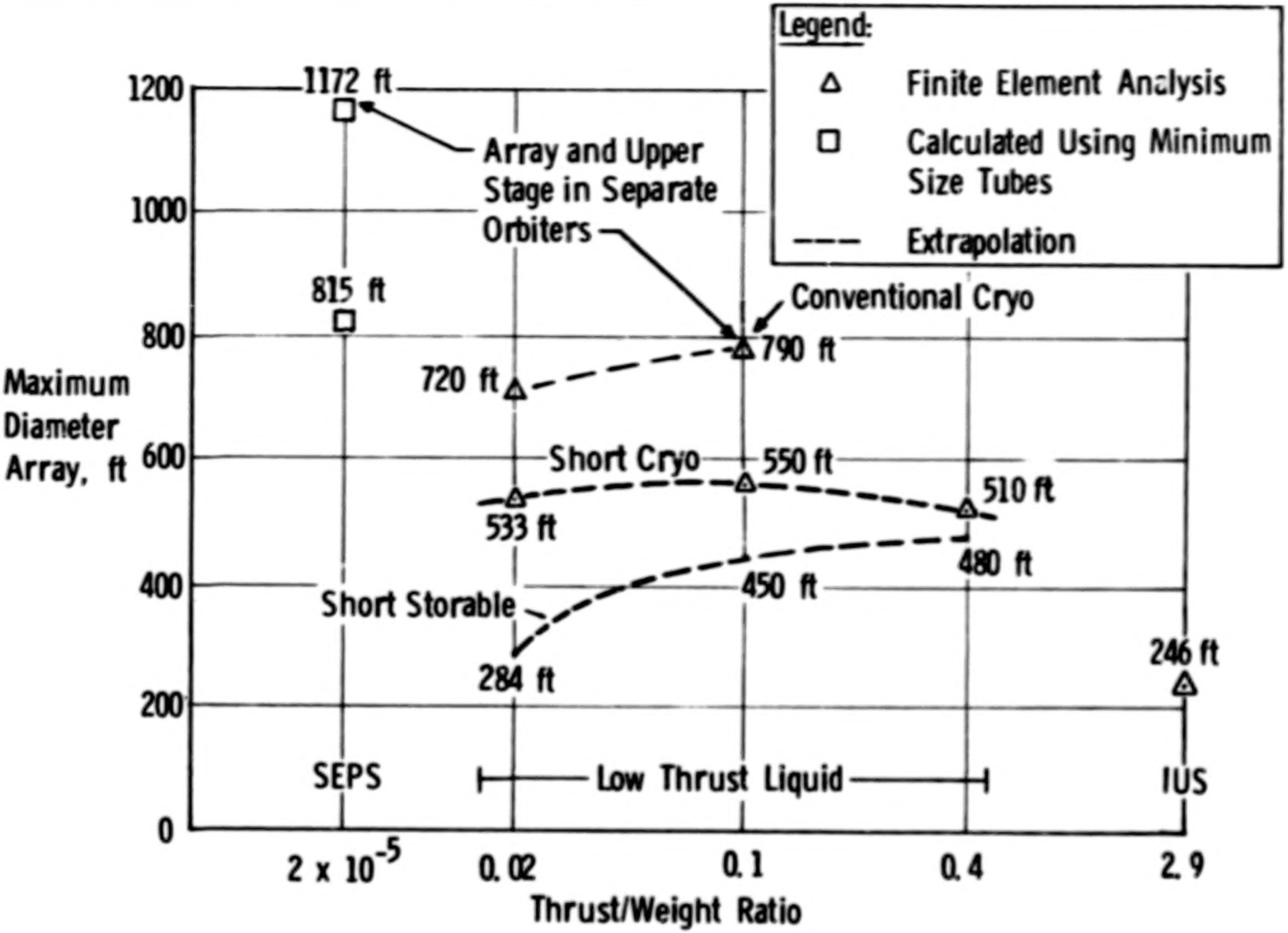


Figure 15

399

FUTURE ASSEMBLY ACTIVITIES (Figure 16)

The next immediate activities associated with the On-Orbit Assembly efforts are the upper stage selection by SAMSO, and the General Dynamics design concept selection by SAMSO.

Once these selections are made, the contractors will begin their Preliminary Concept Design efforts (Feb 78) for the 600-foot baseline spacecraft. This effort will lead to the second program milestone review (May 78).

Examples of additional system requirements would be those associated with on-orbit assembly applications to communications or surveillance systems.

Further program activity will depend on USAF and DDR&E guidance.

All NASA and DOD personnel are invited to follow the On-Orbit Assembly program by attending the periodic program reviews. Those interested should contact Lt.Col. Frank Woods or Captain Paul Heartquist (213) 643-0633 or FTS: 793-0633.

FUTURE ASSEMBLY ACTIVITIES

- SELECT UPPER STAGE FOR ORBITAL TRANSFER
  / GDC-ANTENNA CONCEPT SELECTION
- CONDUCT 600 FOOT DESIGN
- DEFINE ADDITIONAL SYSTEM REQUIREMENTS
- INITIATE FURTHER PROGRAM ACTIVITY

Figure 16

401

402

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DEPLOYABLE ANTENNA DEMONSTRATION STUDY

NASA/GRUMMAN

CONTRACT NAS8-32394

WILBUR THOMPSON MSFC

JACK SCHULTZ GRUMMAN AEROSPACE

403

404

(Figure 1)

The study objective is to define, as a minimum, a shuttle-attached deployment demonstration to support the future system objectives of all NASA centers. Excellent cooperation and useful study inputs have been provided by all participants.

DEPLOYABLE ANTENNA DEMONSTRATION STUDY

MSFC ORGANIZATION: PROGRAM DEVELOPMENT PS04

HQS PROGRAM OFFICE: OSF/MTE

CONTRACTOR: GRUMMAN AEROSPACE CORPORATION

PURPOSE: TO DEFINE A DEMONSTRATION SYSTEM AND SHUTTLE FLIGHT PROGRAM TO DEMONSTRATE PACKAGING, TRANSPORTATION, ERECTION, AND STRUCTURAL INTEGRITY OF A LARGE DEPLOYABLE ANTENNA CONCEPT.

OBJECTIVES:

- ANALYSIS OF DEPLOYABLE ANTENNA REQUIREMENTS
- IDENTIFY AND EVALUATE CANDIDATE CONFIGURATIONS (50-150 METER) OF VARIOUS TYPE ANTENNAS FOR ON-ORBIT DEPLOYMENT FROM THE SHUTTLE
- DEFINE A DEMONSTRATION SYSTEM, OR SYSTEMS, JOINTLY SELECTED BY THE CONTRACTOR AND NASA
- DEFINE A DEMONSTRATION FLIGHT PROGRAM INCLUDING COSTS AND SCHEDULES

STATUS: CONTRACT START ON 24 JUNE 1977, FINAL REPORT 31 MARCH 1978.

Figure 1

406

(Figure 2)

The program being defined by the current study will provide data that will support future systems applications of reflectors and arrays.

CONCEPT DEVELOPMENT GUIDELINES

- PROVIDE DEMONSTRATION RESULTS APPLICABLE TO:
  - 3 ANTENNA CONFIGS; REFLECTORS, BOOTLACE LENS, PHASED ARRAY
  - ANTENNA SIZES BETWEEN 50 AND 200 METERS
  - FREQUENCIES BETWEEN 225 MHz AND 14 GHz
- MAKE CONFIGURATION RETRIEVABLE
- COVER A RANGE OF TEST OBJECTIVES
  - DEMONSTRATE LAUNCH, TRANSPORTATION & DEPLOYMENT WITH SHUTTLE
  - STRUCTURAL MEASUREMENTS
  - ELECTRICAL MEASUREMENTS
  - DEMONSTRATE LIMITED MISSION APPLICATION

Figure 2

(Figure 3)

The two general approaches to space structures larger than the payload bay are (1) to launch structural elements (or build them in space) and assemble the structure in space, or (2) to launch a deployable structure. The second approach is the only one that has been used to date. Both approaches certainly will be used, often together, for future systems; both need new technology. The subject study is considering deployable antennas only.

APPROACHES TO LARGE SPACE STRUCTURES (LARGER THAN THE PAYLOAD BAY)

- LAUNCH STRUCTURAL ELEMENTS
  - – ERECT STRUCTURE IN SPACE
- BUILD STRUCTURAL ELEMENTS IN SPACE
  - – ERECT STRUCTURE IN SPACE
- LAUNCH DEPLOYABLE STRUCTURE
  - – DEPLOY (UNFURL) IN SPACE

☐ ALL LARGE SPACE STRUCTURES TO DATE HAVE BEEN DEPLOYABLE
  - – ATS-6
  - – SKYLAB SOLAR ARRAY

Figure 3

(Figure 4)

At this time (January 1978) the major study activity is preliminary design of deployable paraboloids and lenses. If possible, a common deployable structure will be used for either type. The system will also be retrievable for modification and reflight.

DEPLOYABLE ANTENNA LAUNCH CONFIGURATION

100 M DIA ANTENNA

- LENGTH – 35 FT (10.67 M)
- DIAMETER – 6 FT (1.83 M)
- WEIGHT – 4000 LB (1818 KG)

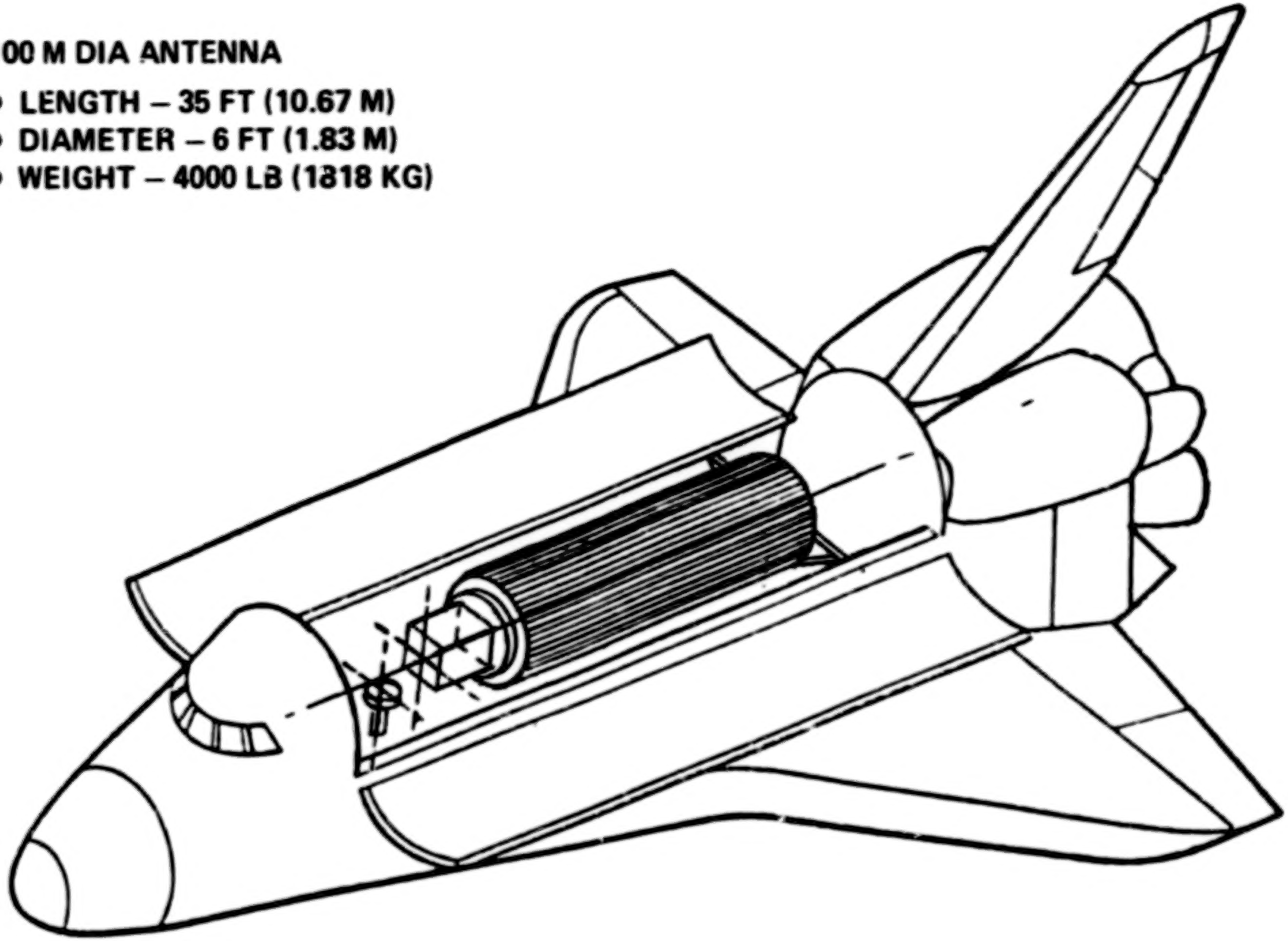


Figure 4

412

(Figure 5)

The baseline configuration is a 100 meter diameter reflector or lens.

CANDIDATE DEMONSTRATION CONFIGURATION 0 – BASIC RIM STRUCTURE

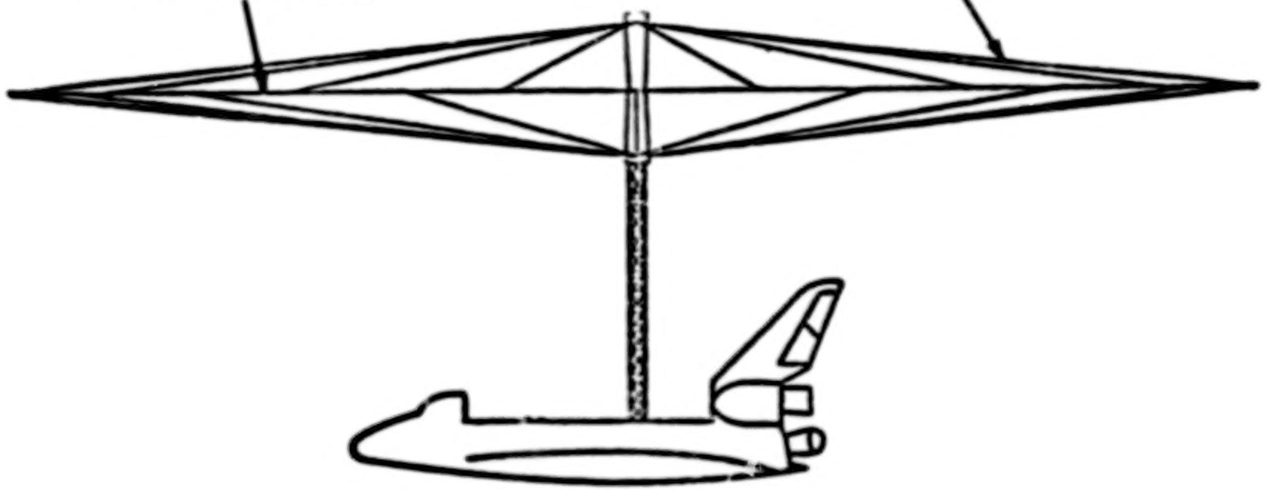


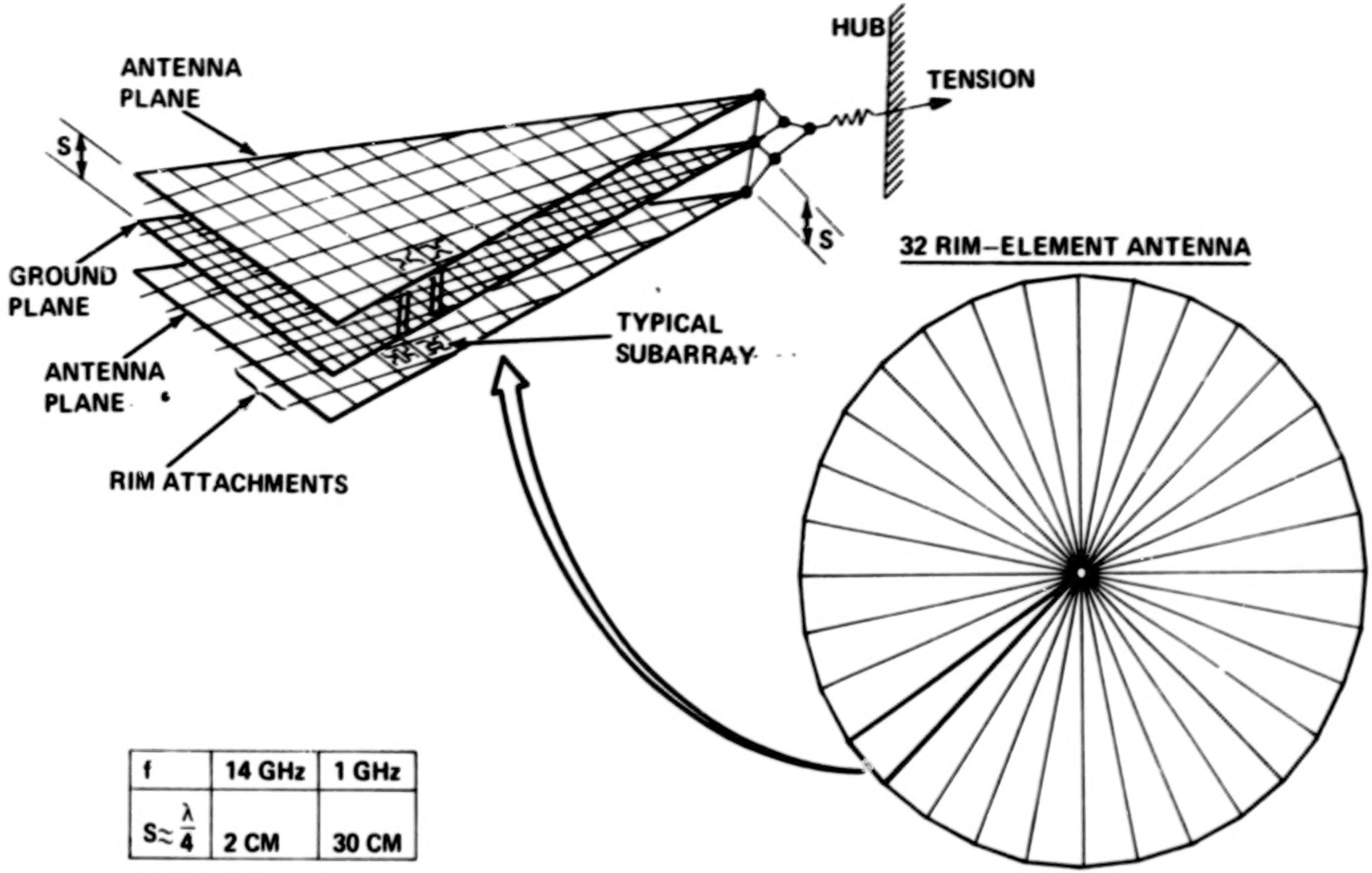
Figure 5

414

(Figure 6)

The phased array and lenses are three parallel planes, a central ground plane and two antenna planes. Bootlace lines through the ground plane connect subarrays on each antenna plane to corresponding subarrays on the other plane. If RF amplifiers are used in the bootlace lines, the system becomes an active lens. If adjustable phase shifters are added, the system becomes an electronically steerable phased array.

TYPICAL GORE ARRANGEMENT FOR PHASED ARRAYS



f	14 GHz	1 GHz
$S \approx \frac{\lambda}{4}$	2 CM	30 CM

Figure 6

416

(Figure 7)

The antenna types in boxes are being considered in the deployable antenna study. The performance limitations of each type are being considered.

LARGE ANTENNA FAMILY TREE

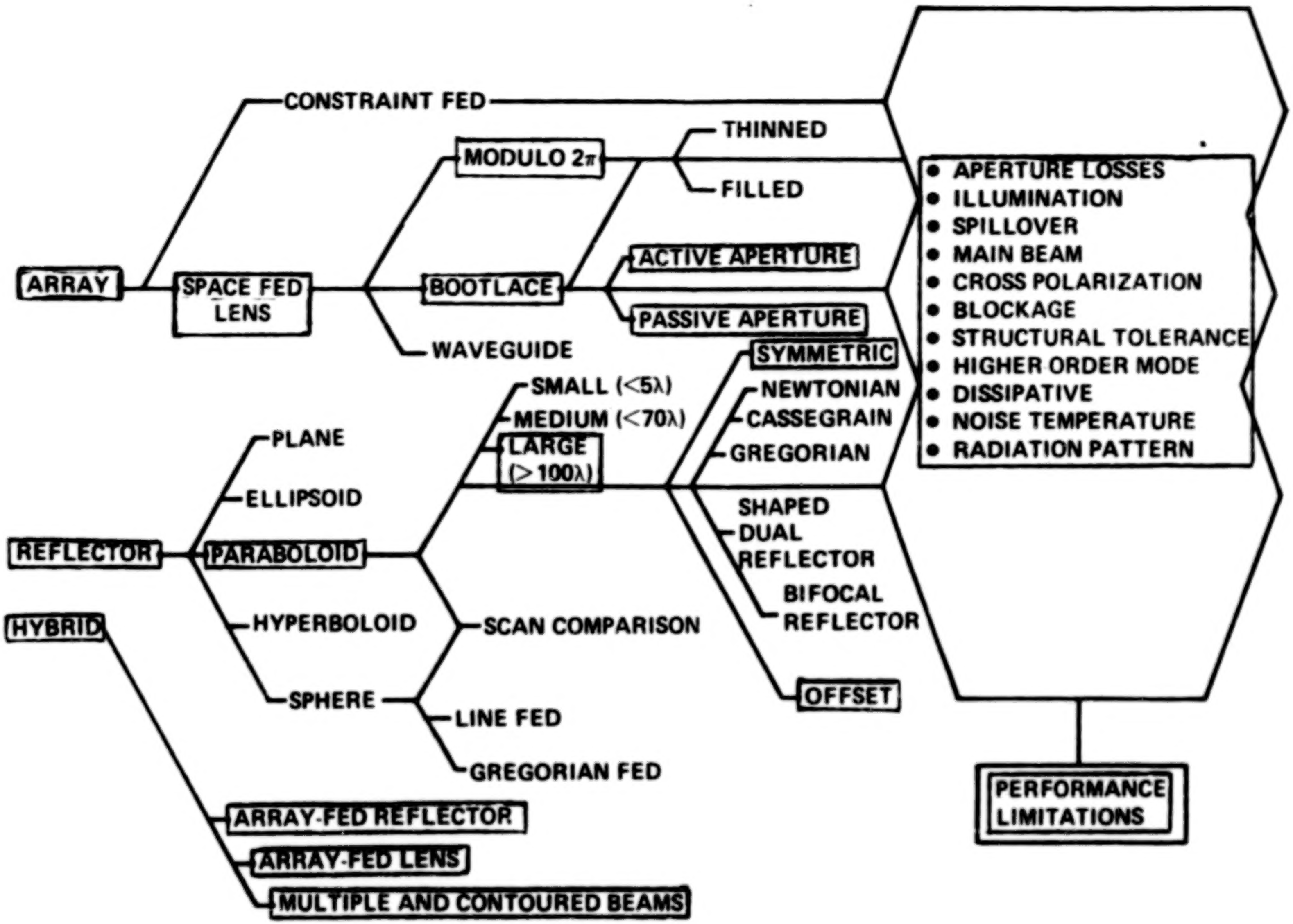


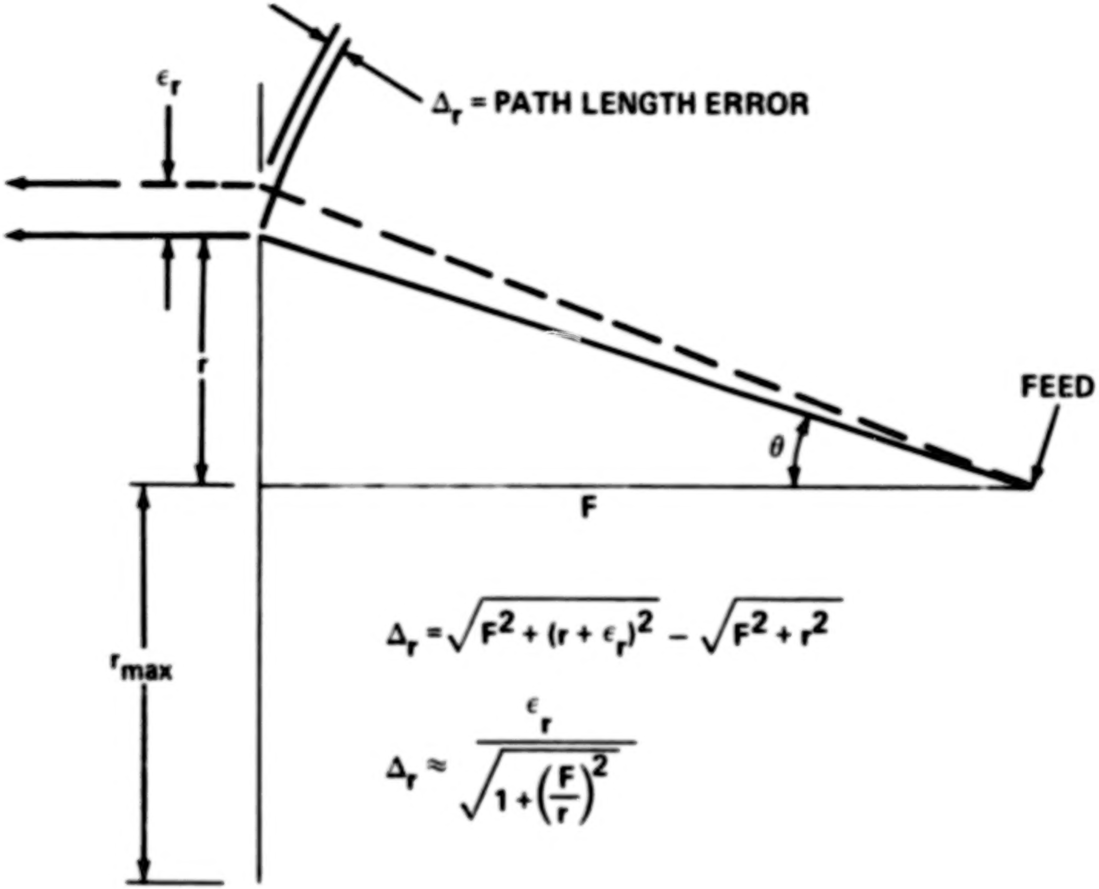
Figure 7

(Figure 8)

Space fed lenses or phased arrays, composed of multiple radiating elements, must consider two tolerances: radial and axial. Radial is defined here.

SURFACE DISTORTION GEOMETRY

FLAT LENS/PHASED ARRAY – RADIAL ERRORS



$$\Delta_r = \sqrt{F^2 + (r + \epsilon_r)^2} - \sqrt{F^2 + r^2}$$

$$\Delta_r \approx \frac{\epsilon_r}{\sqrt{1 + \left(\frac{F}{r}\right)^2}}$$

Figure 8

GRUMMAN

(Figure 9)

The axial tolerance of elements of a lens or phased array is defined here. Comparison with the previous chart will suggest, correctly, that the axial tolerance is much looser than the radial tolerance.

SURFACE DISTORTION GEOMETRY

FLAT LENS/PHASED ARRAY – AXIAL ERRORS

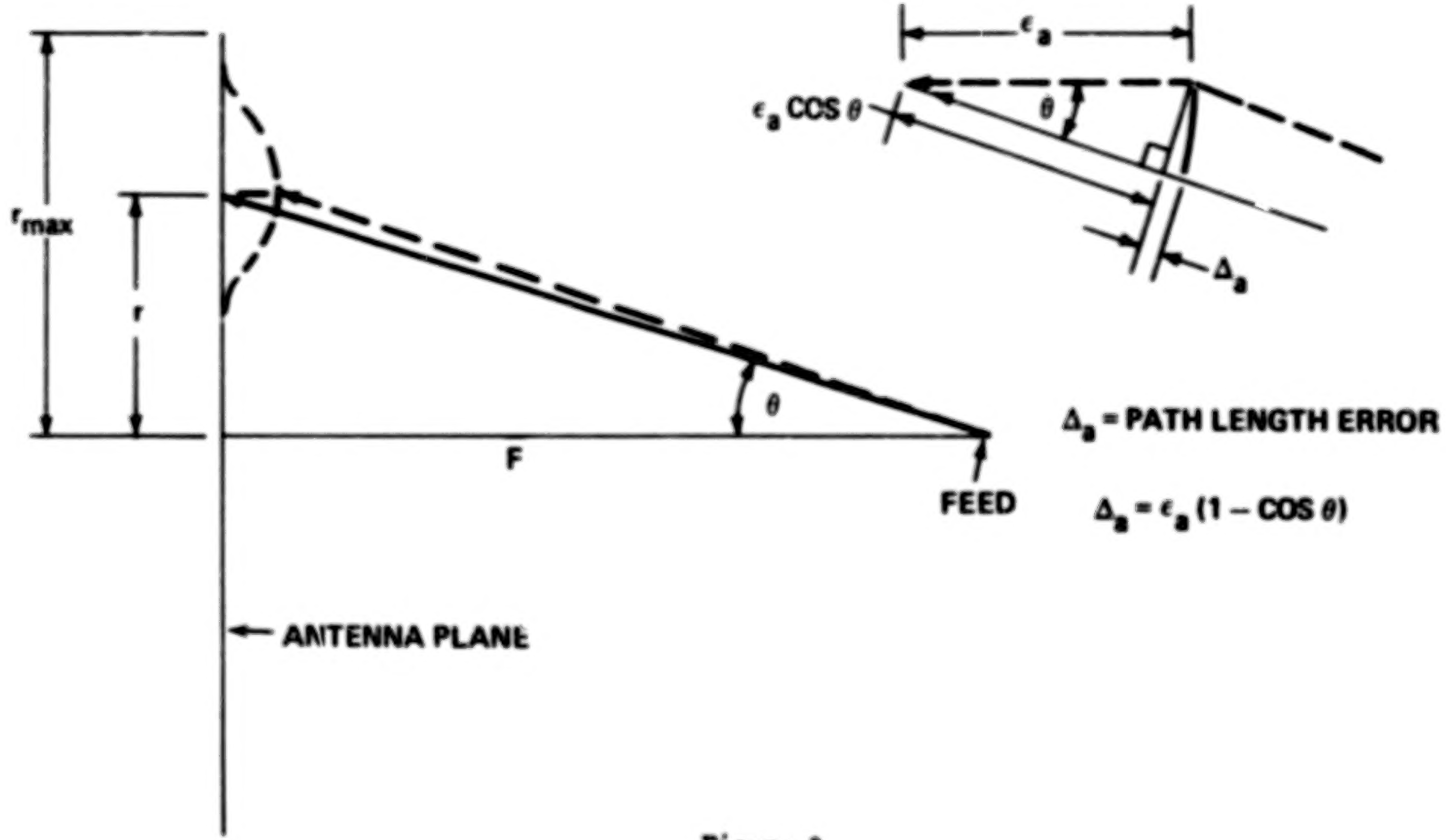


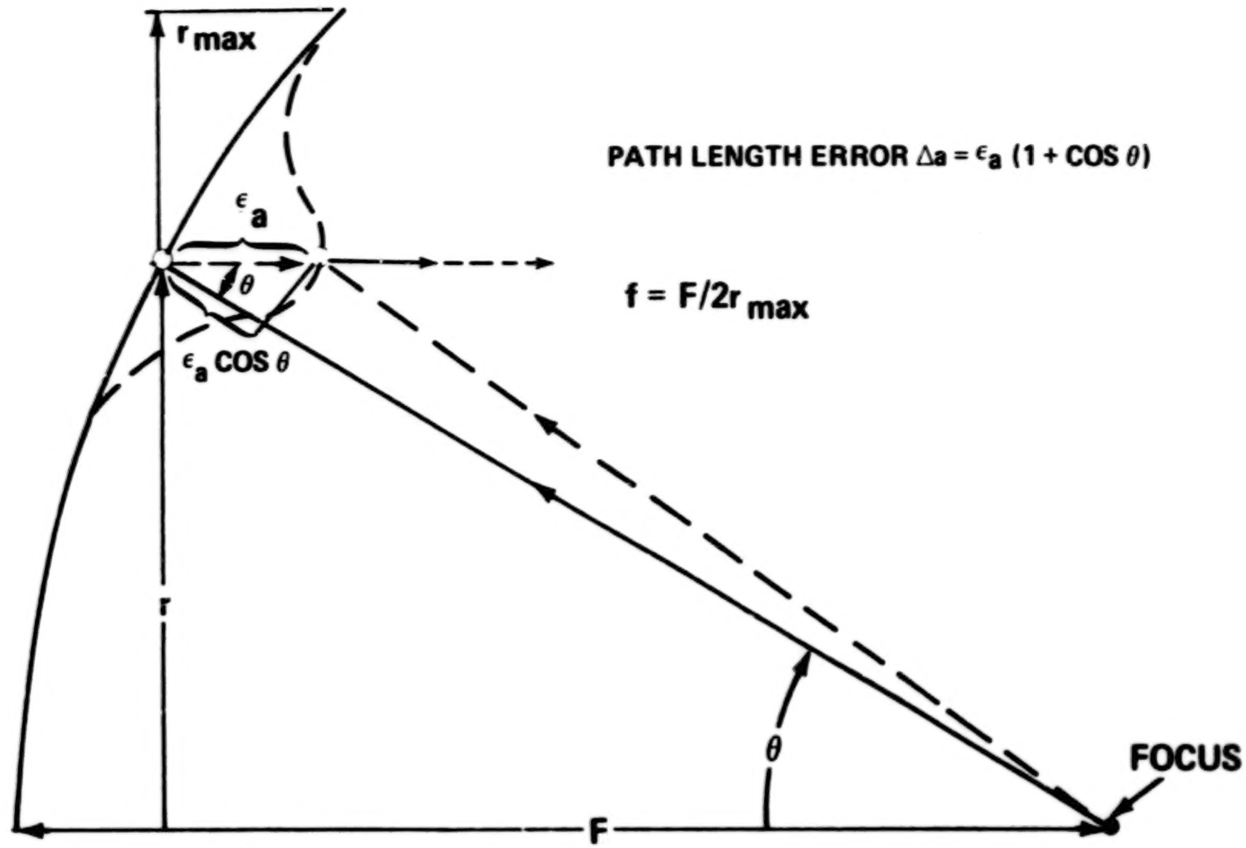
Figure 9

CONTENTS

(Figure 10)

A "bump" in a reflector surface will produce a nearly double-amplitude "bump" in the reflected wave. The path length error for a reflector is defined here.

SURFACE DISTORTION GEOMETRY – REFLECTORS



AXIAL DISTORTION

Figure 10

424

(Figure 11)

A flat-faced space-fed lens or phased array requires that the elements be held to close tolerance in the radial direction. The out-of-plane tolerance is larger than the radial tolerance by a factor of four, if the focal ratio f is one. The axial tolerance is proportional to f^2. The radial tolerance is proportional to f.

STRUCTURAL TOLERANCES – LENS/PHASED ARRAY

TWO COMPONENTS, AXIAL (δa) AND RADIAL (δr)

$$\epsilon_a = \frac{4}{\pi} \lambda \, \sigma \, f^2 \left(\frac{r_{max}}{r} \right)^2$$

$$\epsilon_r = \frac{\lambda}{\pi} f \, \sigma \left(\frac{r_{max}}{r} \right)$$

λ = WAVELENGTH

σ = ELECTRICAL PHASE ERROR, RADIANS

$$f = \frac{F}{D} = \frac{\text{FOCAL LENGTH}}{\text{ANTENNA DIAMETER}}$$

$$\frac{r_{max}}{r} = \frac{\text{RADIUS OF ANTENNA APERTURE}}{\text{RADIAL DISTANCE LESS THAN } r_{max}}$$

ASSUME ELECTRICAL PHASE ERROR 45°, $f = 1$, $\lambda = 15$cm (2 GHz), THEN

$$\epsilon_{a_{max}} = \frac{4}{\pi} (15)\,(.785) = 15\text{CM}$$

$$\epsilon_{r_{max}} = \frac{15}{\pi} (.785) = 3.75 \text{ CM}$$

CONCLUSION: RADIAL TOLERANCE MORE CRITICAL THAN AXIAL TOLERANCE BY A FACTOR OF FOUR

Figure 11

(Figure 12)

The axial tolerance of a reflector must be held closely. For an f-one system, the lens axial tolerance is 16 times larger than the reflector axial tolerance.

STRUCTURAL TOLERANCES – PARABOLIC REFLECTOR

ONE COMPONENT, AXIAL

$$\epsilon a = \frac{\sigma\lambda}{4\pi}$$

FOR SAME ASSUMPTIONS, $\epsilon a = \frac{.785}{4\pi}\ \lambda = \frac{\lambda}{16} = \frac{15}{16} = 0.94\ \text{CM}$

SUMMARY

- LENS/PHASED ARRAY MOST SENSITIVE TO RADIAL ERRORS WHILE THE REFLECTOR IS MOST SENSITIVE TO AXIAL ERRORS
- FOR THE EXAMPLE, THE REFLECTOR STRUCTURAL TOLERANCE IS 16 TIMES MORE STRINGENT COMPARED TO THE LENS/PHASED ARRAY

Figure 12

(Figure 13)

Structural design studies indicate that lightweight deployable phased arrays or lenses can hold tolerance within 8 parts per million units of diameter, permitting the antenna gain to be as high as 84 dB. The diameter or frequency can therefore be anything up to the limit shown.

Using comparable technology for mesh deployable parabolic reflectors should permit the gain to be as high as 60 dB, and the diameter or frequency to be anything up to the limit shown.

ANTENNA DIAMETER VS FREQUENCY

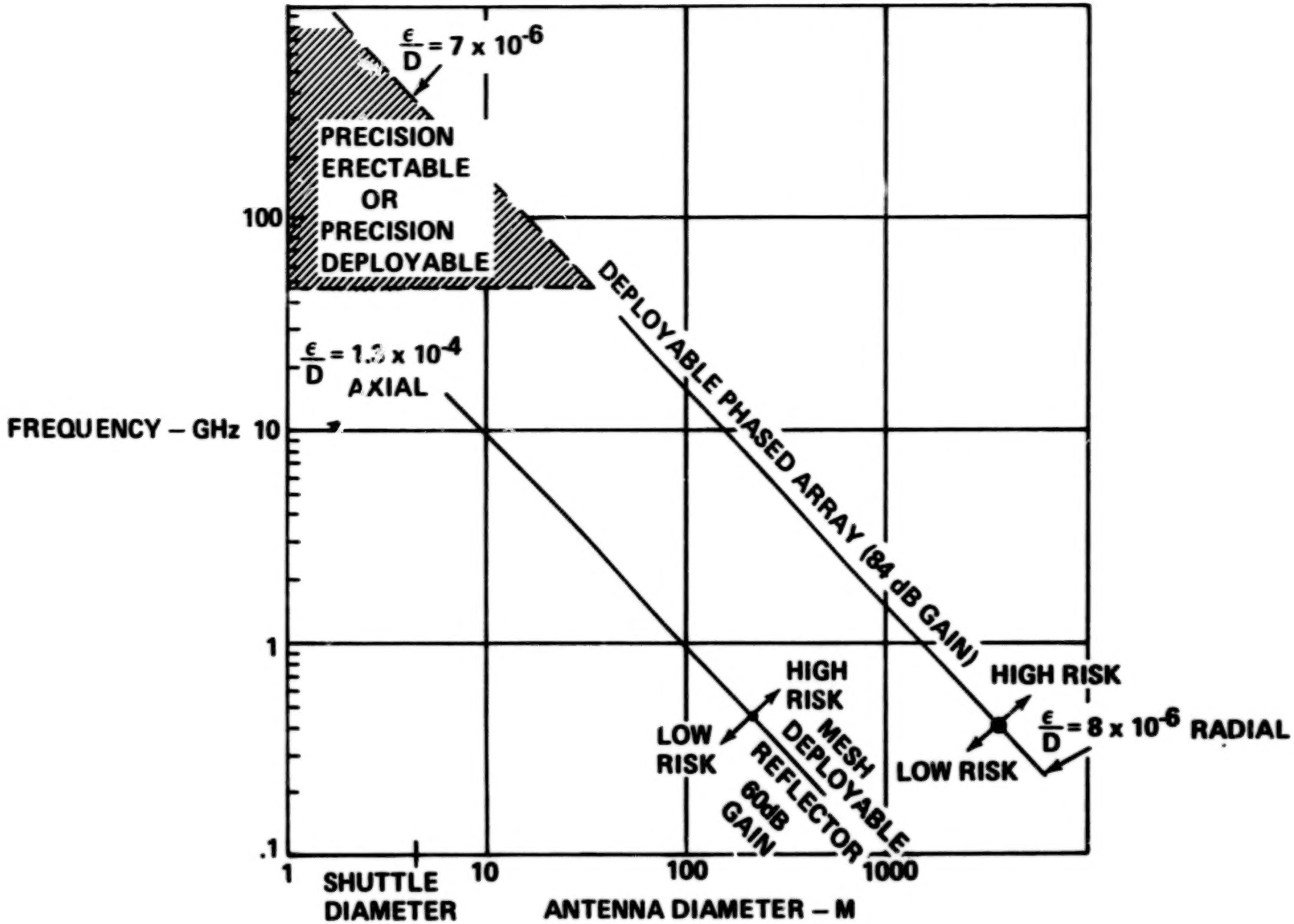


Figure 13

(Figure 14)

Frequency bands allocated to satellite communication services cover a wide range of frequencies. However, the bandwidth as a percentage of the center frequency is relatively narrow. Satellite antennas do need wide instantaneous bandwidth, such as 500 MHz at 11 and 14 GHz.

FREQUENCY ALLOCATIONS FOR SATELLITE COMMUNICATIONS

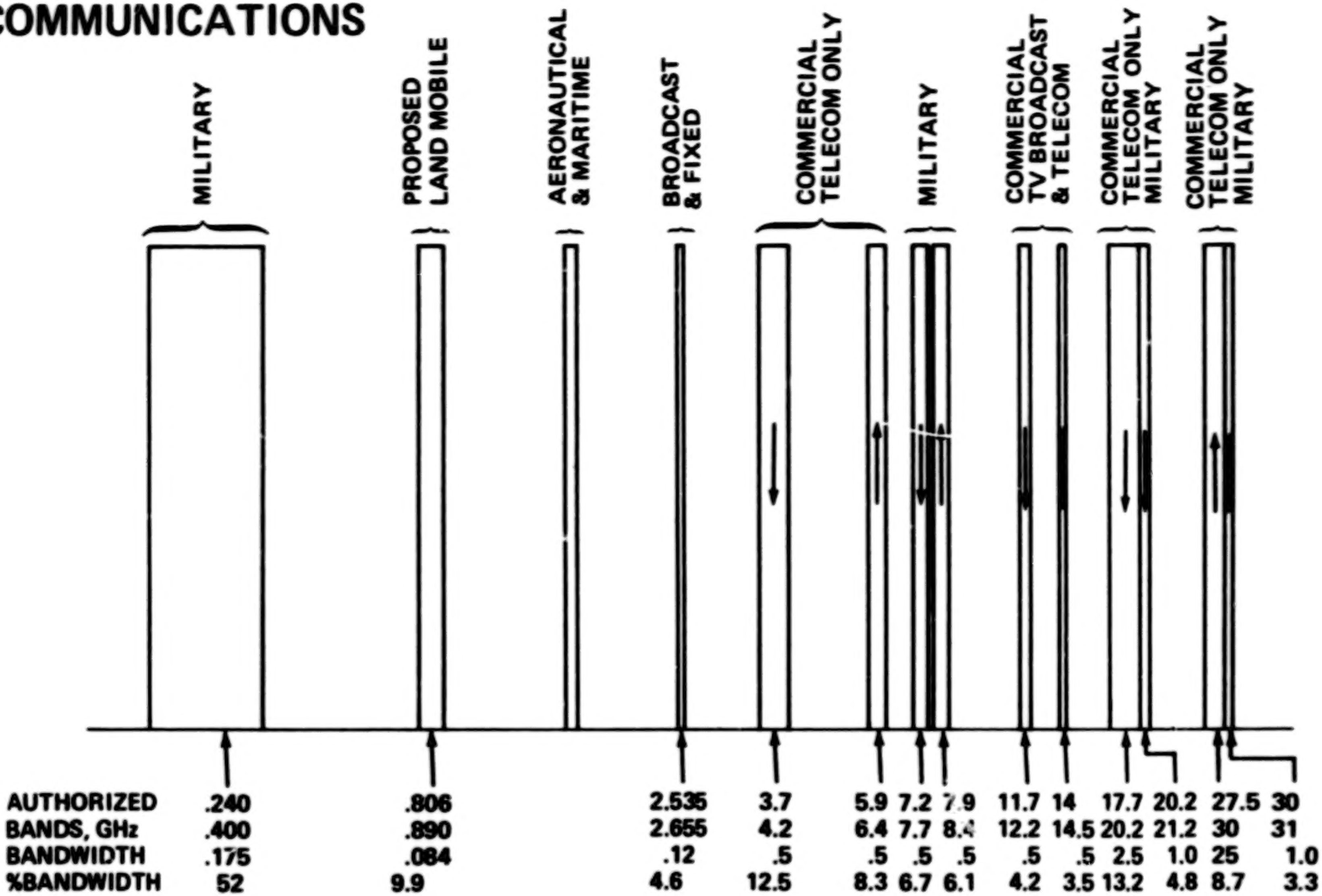


Figure 14

(Figure 15)

The geostationary orbit is a vanishing resource. Discrimination between communication satellites is provided by large ground antennas. Satellites on a common frequency band must be spaced at least 2 degrees.

NEED FOR LARGE ANTENNAS

PRESENT SATELLITE COMMUNICATION

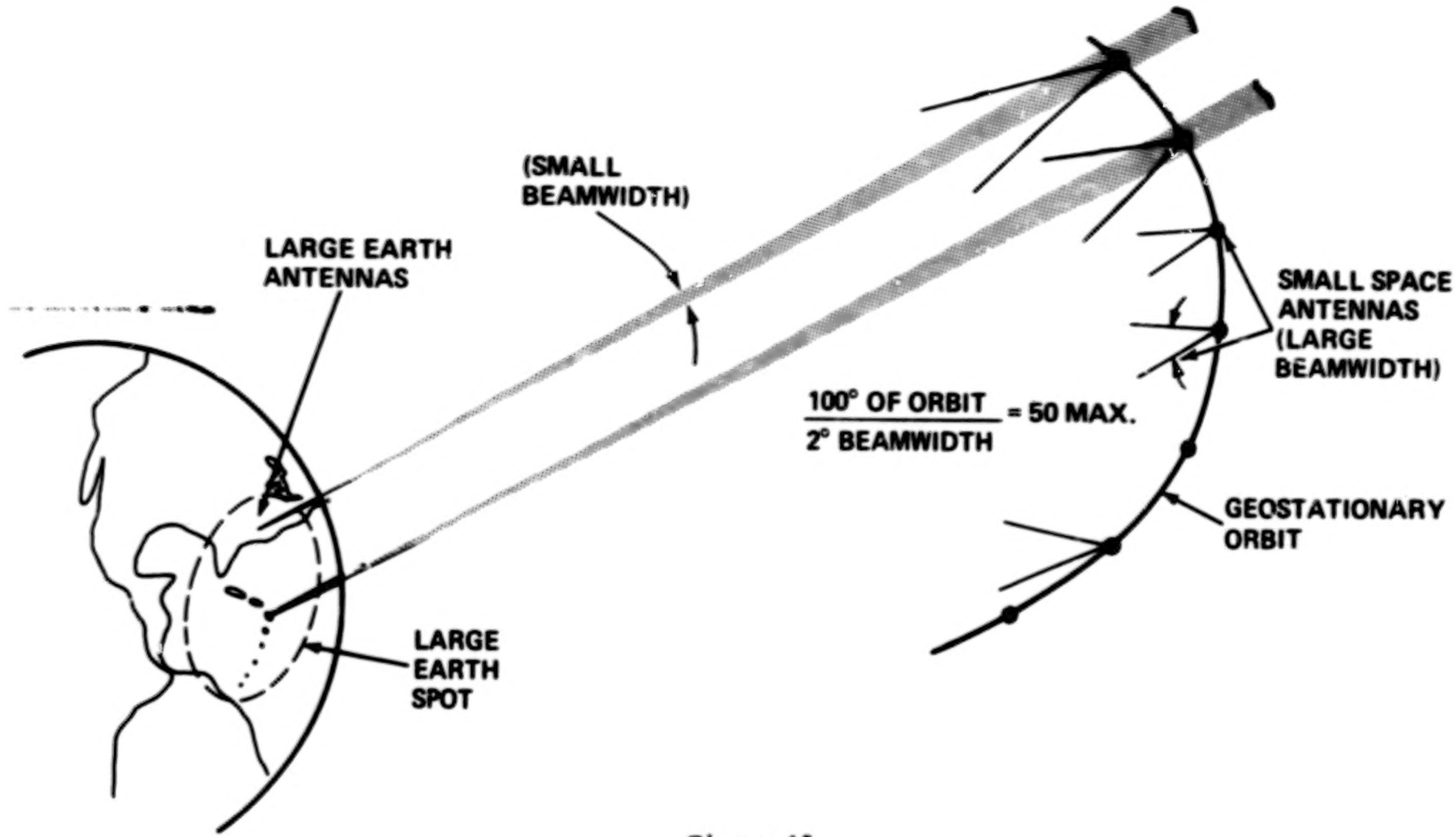


Figure 15

(Figure 16)

Large space antennas, that produce small footprints, discriminate between earth stations. Multibeam space antennas are needed to provide communication links between multiple earth stations; but there is no danger of running out of geostationary parking places; and far more communication channels can be provided by each satellite.

NEED FOR LARGE ANTENNAS

FUTURE SATELLITE COMMUNICATION

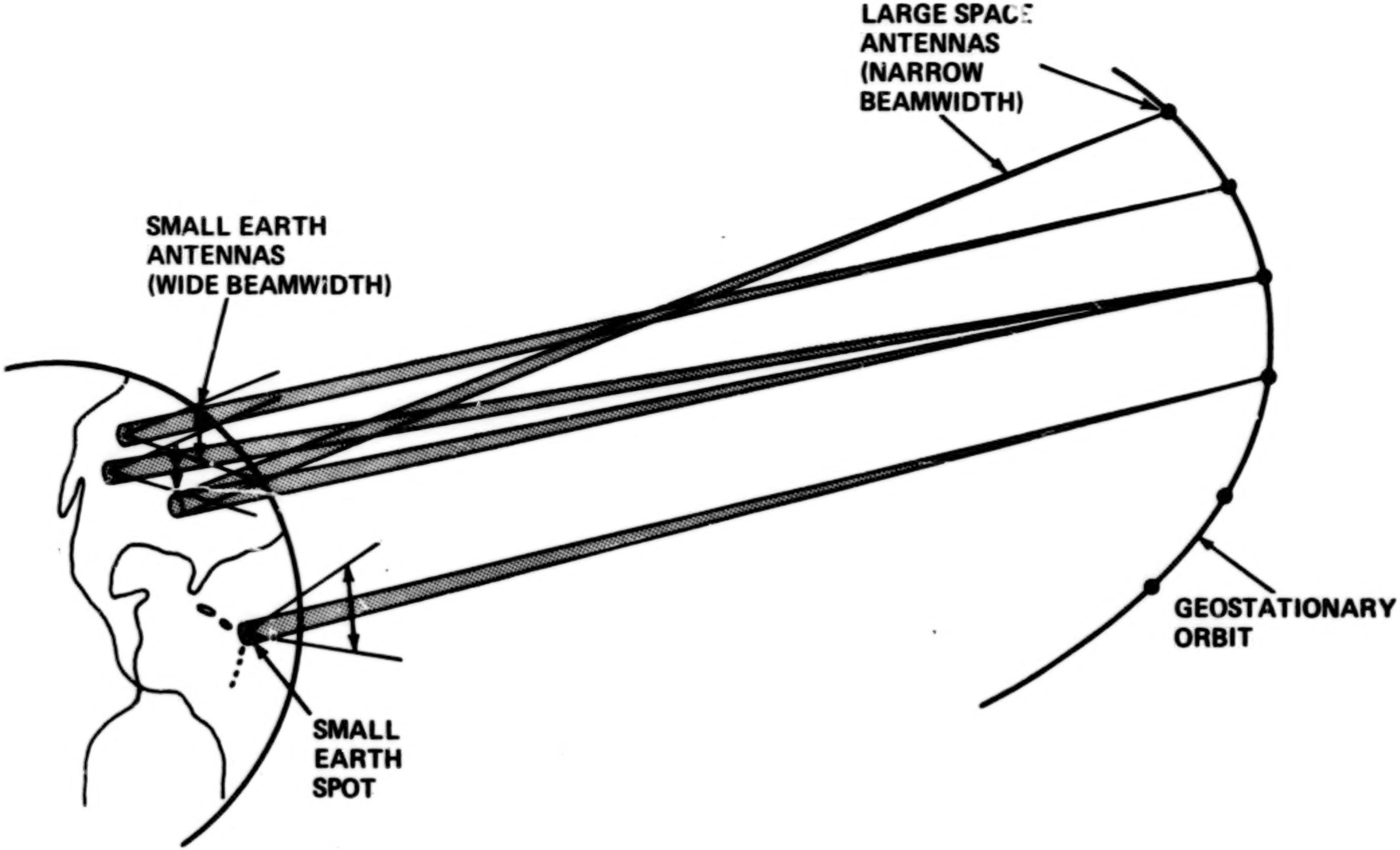


Figure 16

436

(Figure 17)

The deployed antenna system will be nearly as large as the largest ground-based antennas, but more accurate, capable of higher gain and lower sidelobes. The objective of the flight program is to measure the accuracy, gain, and sidelobe performance.

TEST OBJECTIVES

STRUCTURAL

- DEMONSTRATE DEPLOYMENT
- MEASURE DEPLOYED STRUCTURAL ACCURACY
- MEASURE DYNAMIC & THERMAL RESPONSE
- DEMONSTRATE RETRIEVAL

ELECTRICAL

- RF PATTERN AND GAIN MEASUREMENT
- DEMONSTRATE MULTIPLE BEAMS
- ON-AXIS AND OFFSET PARABOLOID, PASSIVE LENS, PHASED ARRAY

APPLICATION

- DEFINE APPLICATION DEMONSTRATION (E.G. RADIOMETRY) FOR LATER FLIGHT

Figure 17

The authors wish to thank J. Bernstein for preparation of a large part of the data presented herein.

SUMMARY

1. ERECTABLE STRUCTURES ARE CLEARLY APPLICABLE TO VERY HIGH PRECISION (ABOVE 50 GHz) ANTENNAS (10 TO 30 M)

2. DEPLOYABLE STRUCTURES ARE LIGHT WEIGHT, APPLICABLE TO MODERATE PRECISION (BELOW 14 GHz) LARGE ANTENNAS (UP TO 300 M)

3. BOTH DEPLOYABLE AND ERECTABLE APPLY TO SMALL AND LARGE ANTENNA FARMS (PLATFORMS WITH COMMON UTILITIES)

4. ERECTABLE AND DEPLOYABLE STRUCTURES NEED DEMONSTRATION IN SPACE

5. LARGE DEPLOYABLE ANTENNA DEMO PROJECT WILL:
   - PROVIDE CAPABILITY TO DEPLOY A LARGE STRUCTURE (30-300 M)
     - PARABOLIC OR FLAT FACE, HIGH F/D
     - MODERATE STRUCTURAL ACCURACY
   - FOR:
     - REFLECTORS, LENSES AND PHASED ARRAYS
   - TO SUPPORT MISSION APPLICATIONS:
     - MULTI-BEAM COMM
     - RADAR
     - RADIOMETRY
     - SPS TARGET
     - SUNLIGHT REFLECTOR

Figure 18

GRUMMAN

440

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PRESENTATION TO GOVERNMENT/INDUSTRY SEMINAR

ON LARGE SPACE SYSTEMS TECHNOLOGY (LSST),

LANGLEY RESEARCH CENTER

TECHNOLOGY FOR ACCURATE SURFACE AND ATTITUDE CONTROL OF A LARGE SPACEBORNE ANTENNA AND MICROWAVE SYSTEM

JOHN. B. DAHLGREN
JET PROPULSION LABORATORY
JANUARY 17, 1978

The idea for this paper came from a study carried out recently at JPL on a Large Spaceborne Antenna and Microwave System with the objective of identifying critical technologies required to enable a large antenna of a few hundred meters in diameter, operating at high frequencies, and having broad user capability (e.g. astronomy, communications, radiometry) toward the end of the century. These capabilities will dictate the need for many new enabling technologies in controls and in other systems. The NASA LSST Program will make a major contribution in satisfying these technology needs.

441

PROBLEM (Figure 1)

The purpose of this paper is twofold: first, it is intended to be provocative and thereby stimulate further thinking and discussion on new technology needs in support of a precision large space system; and second, it illustrates an application which will depend heavily on LSS Technology. The technologies required for the control of a large diameter (200 - 300 m) spaceborne antenna and microwave system operating at frequencies in the range from 20 GHz to at least 300 GHz were reviewed. This review was part of a recent JPL study which had the objective of identifying critical technologies which would be required to enable a system in the 1990 to 2000 time frame. The potential user interest covered broad areas of radio and radar astronomy, orbiting deep space relay station (ODSRS), search for extraterrestrial intelligence (SETI), very long baseline interferometry (VLBI), and earth-looking radiometry. Pointing requirements driven by scientific objectives are 0.1 to 0.2 beamwidths or approximately 0.2 arc sec. Utilization of the full aperture (300 m) at the highest frequency dictates an rms surface accuracy of at least 1/20th of a wavelength or 50 microns. The spectral region of 1 millimeter wavelength and below opens new opportunities in radio and radar astronomy, among others, with this future system capability. An illustration of an early application for submillimeter wavelength investigations will be shown followed by the spaceborne antenna requirements, system level, and control technology challenges identified in the study and finally a summary of some main thoughts to leave with the reader.

JPL

PROBLEM

- CONTROL A LARGE DIAMETER ANTENNA (200 - 300M) OPERATING AT FREQUENCIES FROM 20 GHz TO 300 GHz
  - POINT WITH ACCURACY OF 0.1 TO 0.2 BEAMWIDTHS
  - CONTROL SURFACE ACCURACY TO 50 MICRONS RMS
  - POINT TO ANY NEW TARGET AND SETTLE IN ONE HOUR
  - MINIMIZE DEPENDENCE ON EXPENDABLES
- IDENTIFY THE CRITICAL TECHNOLOGIES REQUIRED TO ENABLE SYSTEM BY 1990 TO 2000 FOR BROAD CLASS OF USERS: RADIO/RADAR ASTRONOMY, ODSRS, SETI, VLBI, AND EARTH-LOOKING RADIOMETRY
- RECOMMEND NASA PROGRAMMATIC APPROACH

Figure 1

JBD-2
1/17/78

SUBMILLIMETER RADIO ASTRONOMY (Figure 2)

The opening of a new spectral region, from 1 to 0.1 millimeter wavelength (in the so called submillimeter range), presents an exciting prospect for radio astronomy. The example illustrates a 10-m antenna with a required surface accuracy of 50 microns being evaluated on the Shuttle. Observations above 300 GHz using a space-based antenna would especially interest radio astronomers because such observations are not accessible from the ground through the atmosphere. A space antenna operating in this spectral region would aid observations of interstellar clouds, extragalactic radio sources, cosmic background, and others. The antenna shown on the Shuttle could be designed along the lines of a ground antenna developed at CalTech by Dr. R. Leighton. Dr. Leighton has been able to fabricate a 10-m parabolic antenna from hexagonal segments in such a way that the surface and the supporting structure can be disassembled for transport, reassembled and yet maintain a surface accuracy of 50 microns.

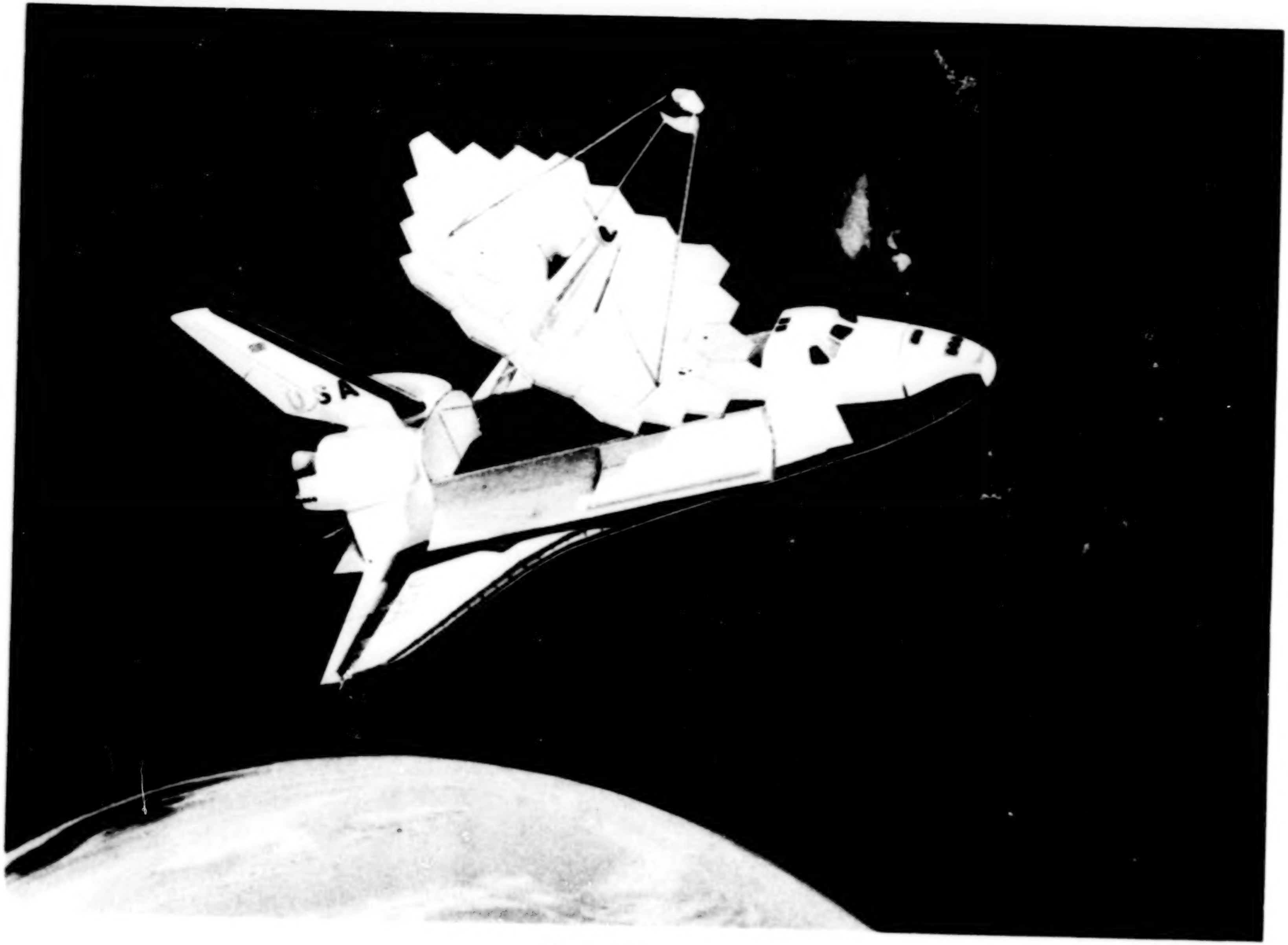

Figure 2

PRINCIPAL SYSTEM FUNCTIONAL REQUIREMENTS (Figure 3)

For a large spaceborne antenna to be cost effective, it should satisfy a broad set of user functional requirements. An operational lifetime of 10 years minimum in GEO was considered necessary to fulfill the goals of most systems. The emphasis on the spaceborne antenna and microwave system technology study was mostly toward outward looking (away from earth) objectives because of the exciting new areas available relative to Earth-based operations. Some examples are radio astronomy applications for all sky surveys, microwave lines investigations, and very long base line interferometry, and in radar astronomy the investigations of asteroids and outer planets. Operating frequencies in the medium to high gigahertz range make these investigations possible outside the earth's atmosphere. Utilizing the full aperture of 300 m at 300 GHz requires surface accuracy control to 1/20 of a wavelength for acceptable operation. With any large space structure system, the design concept must be compatible with the launch vehicle which, in this case, is the Shuttle. The spaceborne antenna and microwave system is clearly an erectable structure and therefore requires multiple launches for transport of parts and materials plus assembly in Shuttle orbit.

PRINCIPAL SYSTEM FUNCTIONAL REQUIREMENTS

- 10 YEAR OPERATIONAL LIFETIME AT GEO
  - RELIANCE ON EXPENDABLES MUST BE MINIMIZED
  - CONSERVATIVE SYSTEM APPROACH REQUIRED
- 20 - 300 GHz (AND HIGHER) OPERATION
  - OUTWARD LOOKING OBJECTIVES EMPHASIZED
  - FREQUENCIES ABOVE 20 GHz NOT ACCESSIBLE FROM EARTH'S SURFACE
- 300M EQUIVALENT APERTURE DIAMETER
  - PRECISION SHAPE CONTROL OF ENTIRE SURFACE REQUIRED
- 100% SKY COVERAGE
  - 4π STERADIAN POINTING
  - DEMANDS ACCURATE POINTING AND TRACKING
- SHUTTLE COMPATIBLE CONCEPT
  - DESIGNED FOR EFFICIENT PACKAGING
  - ERECTABLE IN SHUTTLE ORBIT

447

Figure 3

JBD-4
1/17/78

448

SYSTEM LEVEL TECHNOLOGY CHALLENGES (Figure 4)

The system study revealed a number of critical technologies which are required to enable the spaceborne antenna and microwave system. Low-noise amplifiers operating at front-end temperatures (rf section) near absolute zero are required for investigation of celestial radio sources. Cryogenic fluid storage such as liquid helium at 5°K is presently not feasible for much longer than a year. Electronic beam steering techniques together with mechanical boresight pointing must combine to provide the ultimate pointing accuracy of 0.2 arc sec. Dynamic scan range of the beam must be on the order of 15 beamwidths. Shape control of the entire aperture will involve distribution of thousands of sensors and actuators at control points to provide the rms surface control of 50 microns which is 1/20 wavelength at the highest frequency. The orbit transfer propulsion system must be a low thrust type to minimize the propulsion system weight penalty. Orbit transfer times of approximately 200 days would be characteristic of a 0.2g system. Low-thrust electric propulsion appears to provide the highest payload mass fraction wherein upwards of 70% of LEO mass can be placed in GEO. This mass fraction is important when you consider the large spaceborne antenna mass of 300,000 kg. The main disadvantage of electric propulsion is the power system requirement which must provide an estimated 8 megawatts of power.

SYSTEM LEVEL TECHNOLOGY CHALLENGES

- LOW NOISE AMPLIFIERS AND ASSOCIATED LONG-LIFE CRYOGENICS
  - RECEIVER FRONT-END TEMPERATURES OF A FEW °K
  - CRYOGENIC FLUID STORAGE, E.G., HELIUM AT 5°K
- ARRAY-REFLECTOR ANTENNA TECHNIQUES FOR BEAM-STEERING
  - ULTIMATE POINTING ACCURACY (0.2 ARC SEC)
  - DYNAMIC SCAN RANGE OF 15 BEAMWIDTHS
- SHAPE CONTROL INVOLVING DISTRIBUTED TECHNIQUES
  - SEVERAL THOUSAND CONTROL POINTS
  - RMS SURFACE CONTROL TO 1/20 WAVELENGTH
- SELF-BOOST OF STRUCTURE FROM LEO TO GEO
  - LOW-THRUST (0.2G) ELECTRIC PROPULSION PROVIDES HIGH PAYLOAD MASS FRACTION
    - 70% OF LEO MASS
    - PAYLOAD MASS ESTIMATED AT 300,000 KG
  - LARGE POWER SYSTEM (8 MW_E) REQUIRED

Figure 4

450

SYSTEM LEVEL TECHNOLOGY CHALLENGES (CONTINUED) (Figure 5)

Materials with high strength to weight, high stability, and low thermal coefficient of expansion will be essential in the design of such an antenna. The antenna system mass of 300,000 kg assumed the use of an advanced graphite-epoxy composite material for panels, tubes, trusses, etc., layered in such a way as to produce near zero thermal coefficients. Structural analysis and characterization techniques for large flexible systems must be extended through optimization principles to deal with the large nodal representations. Analysis software to carry out logistics of material, parts, assemblies, tools, etc. to Shuttle orbit must also be optimized because of the implied costs of multiple Shuttle launches. The primary source of loss of surface accuracy in orbit is expected to be caused by uneven sun shading and heating of the structure. These thermal distortions and effects must be understood and will require new and extended thermal analysis tools. Assembly of erectable structures from a Shuttle base is receiving much attention. Utilization of robotic assembly and manufacturing tools will be paramount in the case of the large spaceborne antenna. Multiple Shuttle launches, upwards of 10, will be required to transport the parts and materials to orbit. Thereafter, smart tools and automated operations must be available for efficient assembly.

SYSTEM LEVEL TECHNOLOGY CHALLENGES

(CONTINUED)

- HIGH STRENGTH/WEIGHT, HIGH STABILITY, AND LOW THERMAL COEFFICIENT OF EXPANSION MATERIALS
  - E.G., ADVANCED GRAPHITE-EPOXY LAYUPS
- DESIGN AND ANALYSIS SOFTWARE
  - MULTI-NODE STRUCTURAL OPTIMIZATION
  - LOGISTICS OPTIMIZATION (MATERIAL TRANSPORT)
  - THERMAL ANALYSIS
- ROBOTIC ASSEMBLY OR MANUFACTURING TOOLS
  - SMART TOOLS FOR EFFICIENT ASSEMBLY
  - SPECIAL FABRICATION AND CONSTRUCTION TOOLS

Figure 5

JBD-6
1/17/78

452

CONTROL TECHNOLOGY CHALLENGES (Figure 6)

Shape control was indicated in the system level challenges to be one of the most demanding areas because it involves thousands of actuators and sensors mounted throughout the structure for local control of the surface. The RMS surface deviation allowed from an ideal parabolic reflector at any point is 50 microns or approximately 0.002 inches over the entire aperture of 300 m. Interaction effects of structure on control will also require a distributed system of actuators and sensors in combination with adaptive estimators actively controlling local deformations and interactions. This distributed control concept would be an integral part of attitude and shape control for antenna orientation, configuration, and stabilization. Uncertainties will exist with the antenna control system which cannot be completely determined before flight and therefore adaptive estimators must modify the controller in the presence of nonpredictive frequencies and analytical models. Sensing systems which provide measurement data of the antenna surface and shape to high precision are essential to the system operation. With the Shuttle cargo bay constraints, reflector unit panels are expected to be sized at approximately 15 m^2. For a 300-m parabolic antenna, this will imply some 5000 or more panels. Sensors in combination with actuators mounted at each corner of a panel (and common to one or more panels) would be scanned periodically for conformance. Survey time for the entire surface will be influenced by the finite integration time at each sensor or sensed position. To allow repositioning of the antenna, say 90°, in one hour including settling time and re-tuning of the structure may dictate very rapid survey capability in the order of minutes. Analytic tools for modeling, control system design, and performance verification, etc. of the large antenna represents new areas of technology to deal with the complex integrated systems.

CONTROL TECHNOLOGY CHALLENGES

- SHAPE CONTROL OF REFLECTOR SURFACE CONTROLLED TO 50 MICRONS RMS
  - REQUIRES ≈5 THOUSAND CONTROL POINTS
  - SENSORS AND ACTUATORS DISTRIBUTED THROUGHOUT STRUCTURE
- CONTROL/STRUCTURE INTERACTION EFFECTS CONTROLLED BY DISTRIBUTED ACTUATORS AND SENSORS IN COMBINATION WITH ADAPTIVE ESTIMATORS
- HIGH PERFORMANCE SENSING AND MEASUREMENT SYSTEM
  - ≈5 THOUSAND PANELS INDIVIDUALLY CONTROLLED (PANEL SIZE $\approx 15 M^2$)
  - COMPLETE RE-CALIBRATION AND TUNING REQUIRED WITHIN MINUTES
- ANALYTICAL TOOLS AND TECHNIQUES FOR ANALYSIS OF COMPLEX INTEGRATED SYSTEMS AND TIME EFFICIENT ALGORITHMS

Figure 6

JBD-7
1/17/78

453

454

CONTROL TECHNOLOGY CHALLENGES (Continued)(Figure 7)

Decentralized control through the application of computers and dedicated microprocessors in a partitioned configuration will be required to reduce the complexities of the primary control computer. Integration control concepts for attitude, shape, pointing and interaction will involve multiple distributed actuators and sensors. Torquing devices for attitude and pointing control will involve distribution of momentum/reaction wheels and thrusters working in combination with magnetic control and gravity gradient torquing devices. Multiple precision sensors for celestial and inertial control and for surface measurements must also be integrated into the system.

CONTROL TECHNOLOGY CHALLENGES

(CONTINUED)

- DECENTRALIZED CONTROL
- INTEGRATED, INTERACTIVE, DISTRIBUTED, AND ADAPTIVE CONTROL CONCEPTS
    - MOMENTUM WHEELS, THRUSTERS, SURFACE POSITIONERS, AND DYNAMIC CONTROL ACTUATORS
    - CELESTIAL, INERTIAL, AND SURFACE MEASUREMENT SENSORS
- MOMENTUM CONSERVATIVE SYSTEM CONCEPTS

Figure 7

JBD-8
1/17/78

SUMMARY

- SPACEBORNE ANTENNAS ARE THE ONLY MEANS OF EXTENDING SCIENTIFIC HORIZONS FOR THE MAJORITY OF THE MICROWAVE REGIME

- THE SPACEBORNE ANTENNA AND MICROWAVE SYSTEM REQUIRES EXTREMELY HIGH SURFACE ACCURACY COMBINED WITH PRECISION POINTING

- ATTITUDE AND SURFACE CONTROL SYSTEMS MUST INCLUDE DISTRIBUTED CONTROL ELEMENTS TO ACCOUNT FOR ANTENNA FLEXIBILITY TO SATISFY OVERALL SYSTEM PERFORMANCE

- THE LSST TECHNOLOGY PROGRAM IS KEY TO FUTURE LARGE ANTENNA SYSTEMS

Figure 8

INFLIGHT OPTICAL MEASUREMENT OF ANTENNA SURFACES

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FOR: NASA/DOD/INDUSTRY SEMINAR
LARGE SPACE SYSTEMS TECHNOLOGY
LANGLEY RESEARCH CENTER
JANUARY 17-19, 1978

457

458

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OUTLINE

IN-FLIGHT OPTICAL MEASUREMENT OF ANTENNA SURFACES

- OBJECTIVES
    - REPRESENTATIVE APPLICATIONS
    - APPLICATION DERIVED REQUIREMENTS
- CHOOSING AN OPTICAL MEASUREMENT APPROACH
    - TRIANGULATION
    - COORDINATE DETECTORS FOR TRIANGULATION TECHNIQUE
- DETECTOR CHARACTERIZATION, LATERAL EFFECT SILICON PHOTODIODE
    - DESCRIPTION
    - EVALUATION PROGRAM
        - COMPUTER MODELLING
        - RESPONSE TESTS
    - EXPERIMENTAL RESULTS
- PRELIMINARY CONCLUSIONS

460

Objective (Figure 1)

The objective of this effort is to develop a technology base for a wide variety of applications oriented sensors. Requirements are established by the more demanding applications. Therefore, while this technology may be equally useful for remote attitude measurement of instrument packages, as an aid for mooring and docking, and as a guidance sensor for satellite retrieval and servicing, the most stringent demands come from antenna applications: from requirements for fabrication, assembly, test, surface figure monitoring, and ultimately surface figure active control.

OBJECTIVE

- TO DEVELOP A SENSOR TECHNOLOGY BASE FOR THE MEASUREMENT OF DEFORMATIONS OF LARGE STRUCTURES IN SPACE

REPRESENTATIVE APPLICATIONS

- MEASUREMENT OF EXPERIMENTAL STRUCTURE DEFORMATIONS UNDER THERMAL AND MECHANICAL STRESSES
- CONTINUOUS MONITOR OF SURFACE DISTORTION IN LARGE DEPLOYABLE ANTENNAS
- DYNAMIC MEASUREMENT OF SURFACE DISTORTION IN LARGE ERECTABLE ANTENNAS
- FABRICATION AID AND ASSEMBLY AID FOR CONSTRUCTION OF LARGE ANTENNAS IN SPACE

Figure 1

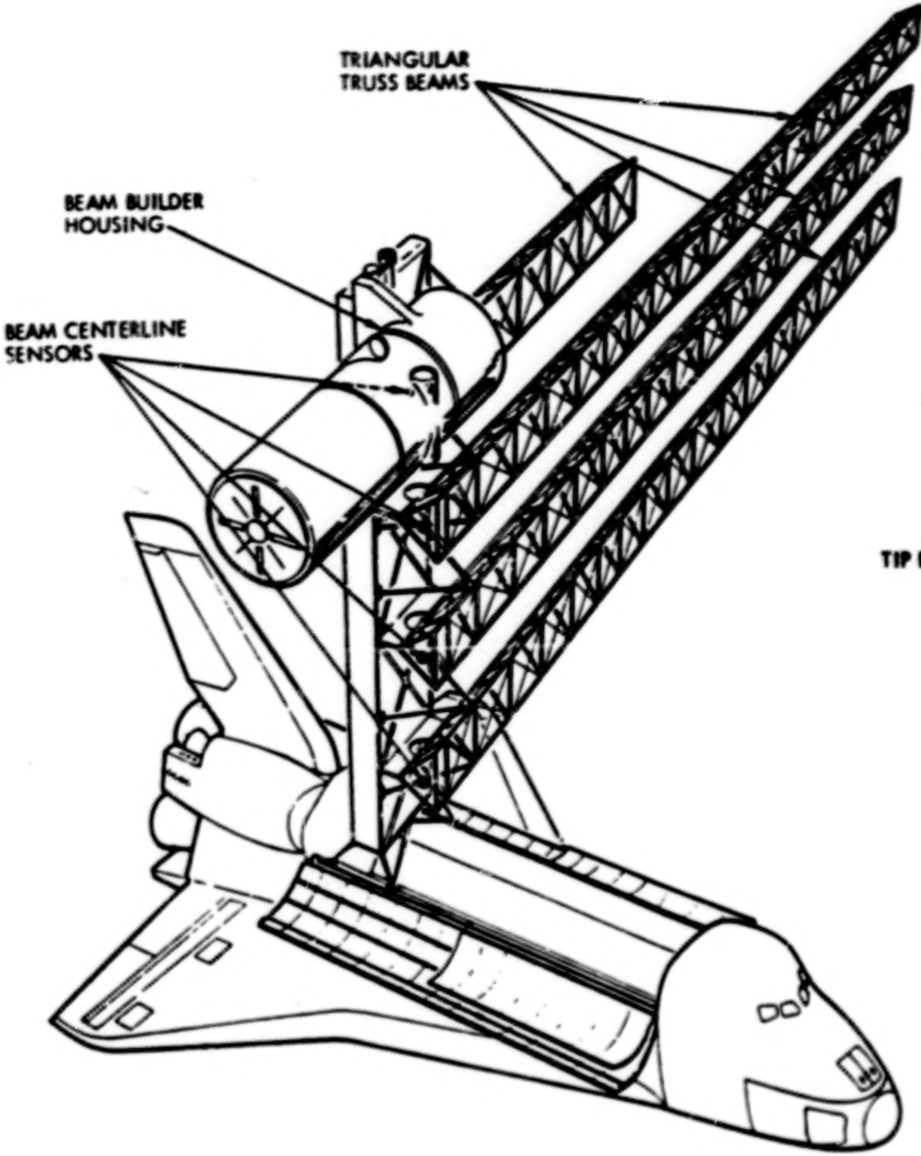
461

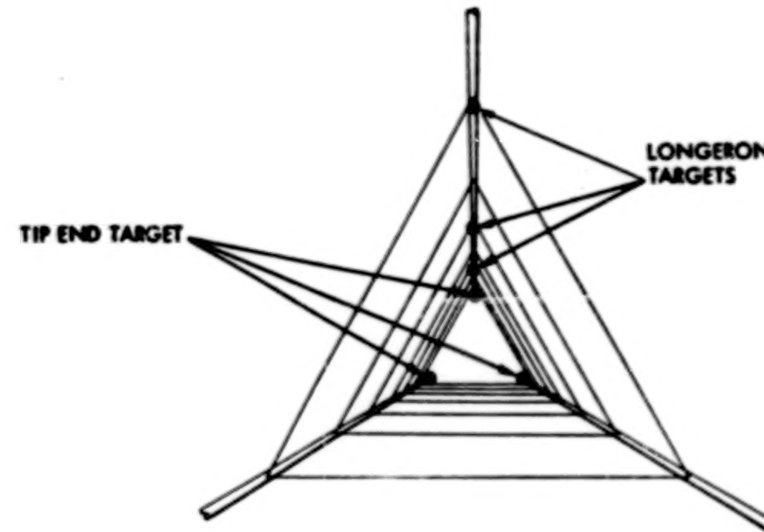
462

Measurement of Beam Deformation During Fabrication and Test (Figure 2)

One approach to achieving large antenna in space is on-site construction and assembly. The manufacturing of composite material beams perhaps 300 to 1000 meters in length will demand extremely precise controls at the beam building machine to prevent undue warping and twisting of the end product. A technique proposed here is an optical sensor establishing an ideal "centerline" at each beam during fabrication or later during assembly. Deviations from the centerline, either in lateral deformation or in twist, are measured to produce limit warnings or to evoke active control at the building machine.

MEASUREMENT OF BEAM DEFORMATION DURING FABRICATION AND TEST





MEASUREMENT:

BEAM BENDING (2 - AXIS)
BEAM TWIST

Figure 2

Measurement of Surface Figure

Large Deployable Mesh Antenna (Figure 3)

Under the assumption that deployable, mesh antenna as large as 100 meters diameter may be practicable, some accurate means of verifying surface figure at deployment and change in figure during orbital cycling is demanded. As many as 500 or 1000 sample points at the surface may be needed to define the antenna directivity or efficiency. A proposed approach is the use of optical sensors at the antenna base surveying arrays of target points situated at tie-line intersections as shown. These targets must not perturb the deployment geometry of the antenna, and they must not affect the rf wavefront.

MEASUREMENT OF SURFACE FIGURE, LARGE DEPLOYABLE MESH ANTENNA

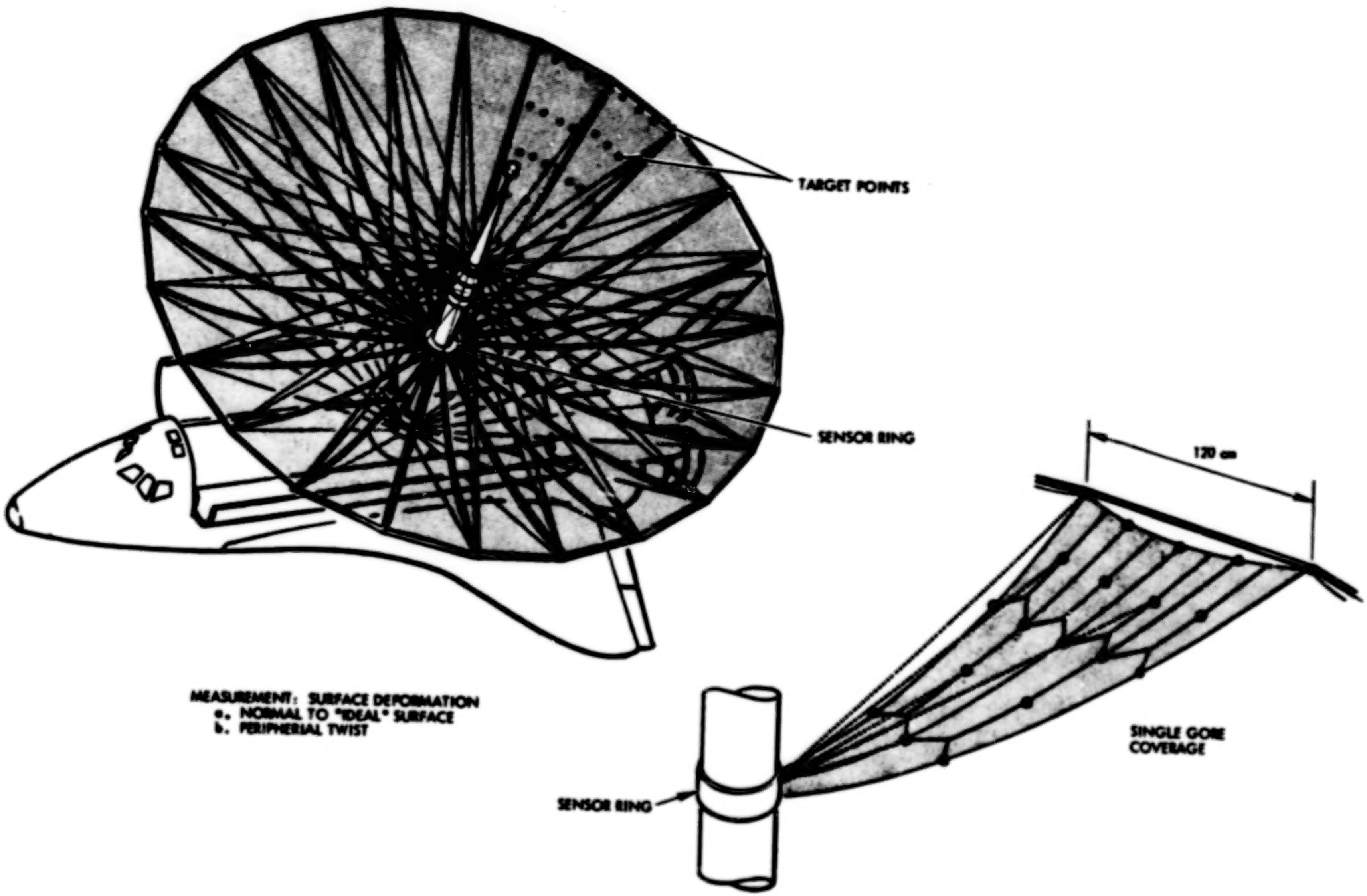


Figure 3

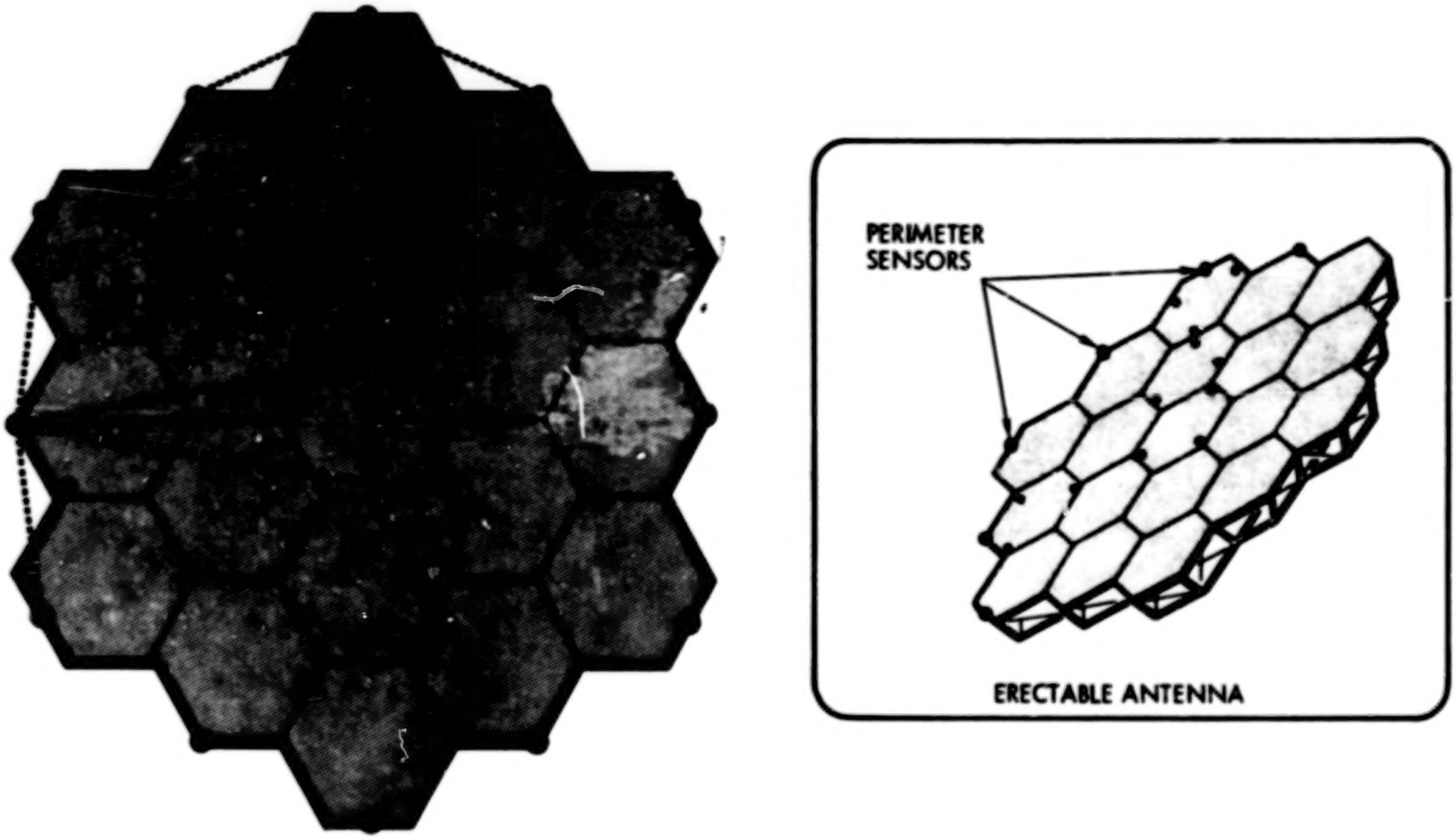
465

Cell Misalignment Sensor System

Large Erectable Antenna (Figure 4)

One method suggested to achieve very large dish or phased array antennas in space is to assemble these from prefabricated elements (such as hexagonal cells). It is unlikely that sufficient structural stiffness can be provided to insure each cell, after installation, will maintain positional and attitude accuracy. Some type of alignment sensing and active control may be required. A schematic of an optical sensing approach is illustrated; the sensors are arrayed around the perimeter of the antenna and maintain dedicated coverage of selected targets and of neighboring sensors. The sensor ring establishes a reference frame from which the attitude and position of each cell is determined. Multiple coverage of the central cell provides a check on the radial axes of the reference frame.

CELL MISALIGNMENT SENSOR SYSTEM
LARGE ERECTABLE ANTENNA



SENSOR COVERAGE, 19-CELL ANTENNA

Figure 4

468

Representative Measurement Requirements (Figure 5)

One of the most critical parameters defining the sensor requirements is the fractional accuracy needed: that is, the absolute accuracy expressed as a fraction of the maximum target excursion. Lineal accuracy is established by factors such as degradation of antenna efficiency or directivity. Maximum excursion to be accommodated by the sensor may be chosen for diagnostics, worst case acquisition, or other application derived demands. For this exercise, the maximum tolerable deviation of an antenna element is taken to be 50 centimeters.

REPRESENTATIVE MEASUREMENT REQUIREMENTS

APPLICATION	TARGET MAX. RANGE	TARGET MAX. EXCURSION[1]	MEASUREMENT ACCURACY (LINEAL)	MEASUREMENT ACCURACY (FRACTIONAL)[2]
EXPERIMENTAL BEAM FABRICATION AND TEST	1000 M (BASE TO TIP)	10 M	10 MM[3]	10^{-3}
DEPLOYABLE MESH ANTENNA FIGURE MONITOR	50 M (COLUMN TO HOOP)	0.5 M	0.5 MM (λ = 30 CM)[4] 50 μM (λ = 3 CM)	10^{-3} 10^{-4}
ERECTABLE CELL ANTENNA MISALIGNMENT SENSOR	500 M (RIM TO CENTER)	0.5 M	50 μM (λ = 3 CM) 5 μM (λ = 0.3 CM)	10^{-4} 10^{-5}

NOTES:

1. MAXIMUM EXPECTED EXCURSION LATERAL TO LINE-OF-SIGHT
2. LINEAL ACCURACY DIVIDED BY THE MAXIMUM EXCURSION
3. EQUIVALENT TWIST MEASUREMENT ACCURACY, 1.5 M BEAM, IS .4°
4. ACCURACY BASED UPON MEDIUM SCALE SURFACE ROUGHNESS OF λ/50.

Figure 5

CONTENTS

VOLUME I

Other Sensor System Requirements (Figure 6)

For many applications, in context with relatively slow deformation rates (typically less than 1 Hz response), continuous target sensing is required. This real-time sensing capability provides for limit warning, diagnostics and remedial actions, and for active control of the antenna elements. Determination of the overall antenna surface geometry from the discrete sampling points requires sophisticated data processing, and thus the sensor system outputs must be fully compatible with on-board processing. For example, to minimize process loads with sensor outputs that require only simple arithmetic, manipulations are preferred over those that demand stored calibration data and complex (eg. multi-term polynomial) corrections.

OTHER SENSOR SYSTEM REQUIREMENTS

- REAL-TIME MEASUREMENT OUTPUTS
- IMMUNITY TO BACKGROUND (SUNLIGHT GLINTS, EARTHSHINE, ETC.)
- MEASUREMENT STABILITY
- COMPATIBILITY WITH SIMPLE REAL-TIME DATA PROCESSING (I.E., LINEAR RESPONSES)
- DIRECT INTERFACE WITH MICROPROCESSORS, FEEDBACK CONTROLLERS AND CONVENTIONAL RECORDERS
- MODULAR SYSTEM ELEMENTS: SIMPLE, RUGGED, INEXPENSIVE (EXPENDABLE IF NECESSARY)
- RELIANCE UPON EXISTING TECHNOLOGY BASE

Figure 6

472

Optical Measurement Approaches (Figure 7)

Although each of the approaches noted has certain ideal applications, most are insufficiently general to be considered for the broad spectrum of uses considered here. Attitude transfer sensors do not measure deflections. Holography and photogrammetry record on film and thereby are not real-time. Lidar can provide ranging with a few millimeter accuracy, but may not be readily extended to give sub-millimeter accuracy. In the triangulation category, servoed beam steering techniques are overly complex. Among the "staring" angle sensors, the choice becomes that of selecting an appropriate detector.

OPTICAL MEASUREMENT APPROACHES

- ATTITUDE ONLY SENSORS
  - MIRROR-PRISM CONFIGURATIONS
  - POLARIZATION TECHNIQUES
- HOLOGRAPHY
- LIDAR (OPTICAL RADAR)
- TRIANGULATION
  - PHOTOGRAMMETRY
  - ANGLE SENSORS
    - SERVOED BEAM STEERING (NULL TECHNIQUE)
    - IMAGE TUBE (IMAGE DISSECTOR, VIDICON, ETC.)
    - CCD ARRAY
    - COORDINATE DETECTOR

Figure 7

474

Triangulation (Figure 8)

The base coordinate frame to which all measurements are referred is at the angle sensor, or receiver. In the receiver-target distance, z, is known, then a single point source (1) can provide a measure of lateral deflections. x and y. If the range, z, is not known, it can be determined by adding a second source (2) at a known spacing from source (1). This also yields twist, θ_z. The addition of the third source, (3) set back in the target, gives the rotations about the lateral axes. Thus, three sources at the target can provide its six degrees of freedom. Many applications, in particular those related to structural deformations, have sufficient constraints so that the full set of six coordinates at each sample point is unnecessary.

TRIANGULATION GEOMETRY

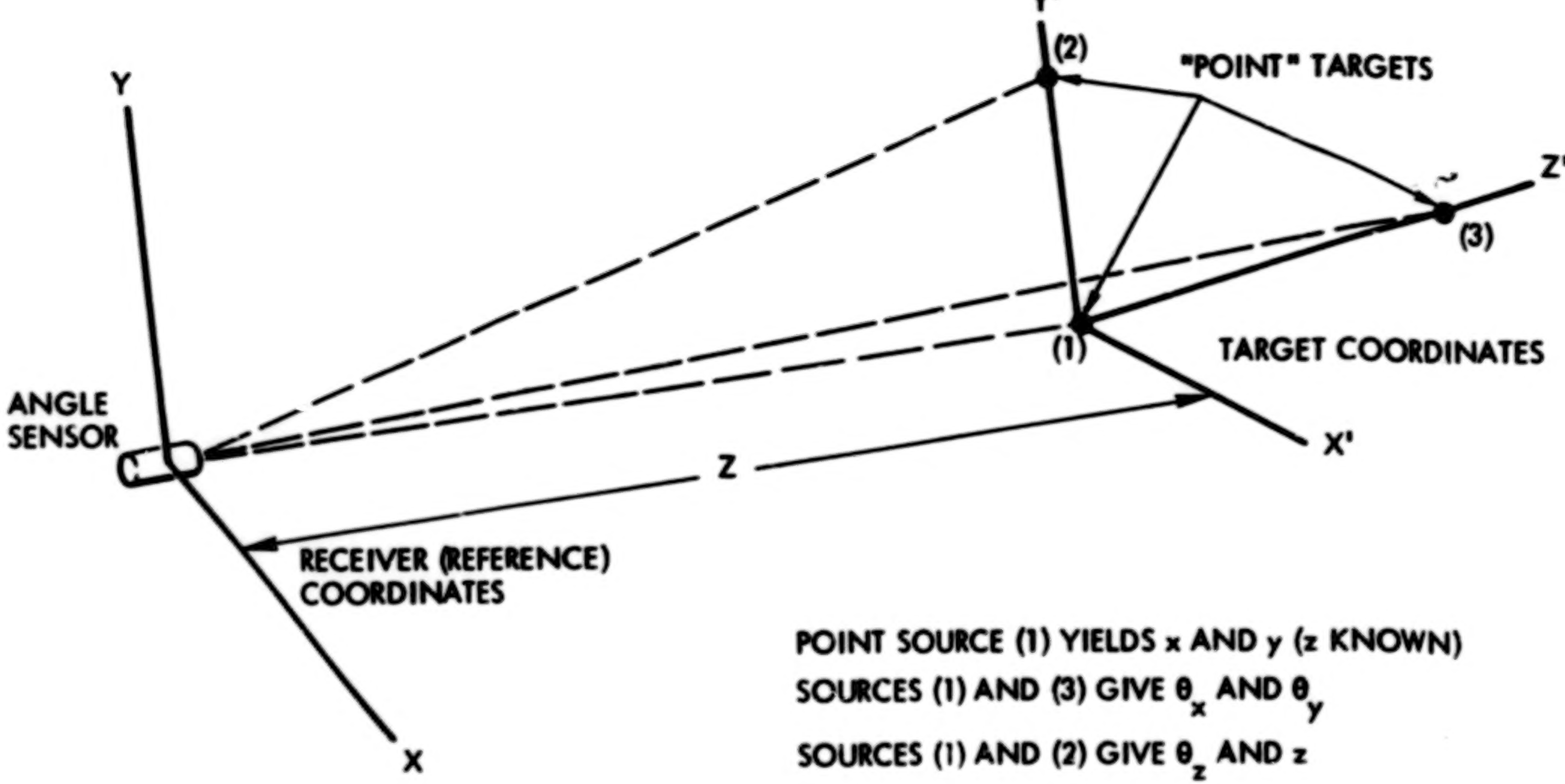


Figure 8

Comparison of Image Position Detectors (Figure 9)

In the triangulation approach, the angle sensing receiver consists of an objective lens and an image position sensor at the focus of the lens. One principal consideration in comparing candidate image position detectors is the fractional or percent accuracy referencing total excursion. A related consideration is the complexity of processing needed to achieve this accuracy. For the return beam vidicon, the accuracies are estimated for bulk processing (fixed transformation) and precision processing (calibrated). Calibrated implies processing corrections unique to individual detectors. For the CCD array, 3x3 or 4x4 cell, fixed processing is assumed with resultant interpolation accuracy of 1/20th of a pixelwidth. Fixed transformation of photodiode signals consists of a simple cubic correction. Although the silicon photodiode is less sensitive than the other detectors, its accuracy, stability, ruggedness, and minimal requirements upon subsequent electronics and processing make it the most favorable selection.

COMPARISON OF IMAGE POSITION DETECTORS

MODE	MAXIMUM ERROR (PERCENT) VIDICON(1) (RBV)	IMAGE DISSECTOR	CCD(2) ARRAY	PHOTODIODE(3) COORDINATE DETECTOR
RAW DATA	2	0.5 TO 1	0.2	0.01 - 0.1
FIXED TRANSFORMATION	0.5	0.1	0.01(4)	0.001 - 0.01
CALIBRATED	0.05	0.03	0.01	< 0.001(5)

(1) DATA FROM SPECIAL RBV FOR LANDSAT INSTRUMENT

(2) ASSUMES 500 BY 500 ARRAY

(3) CURRENT STATUS OF PLANAR DIFFUSE SILICON PHOTODIODES

(4) INTERPOLATION BY CENTROID ANALYSIS OF SPREAD IMAGE

(5) THIS LIMIT HAS NOT BEEN ESTABLISHED

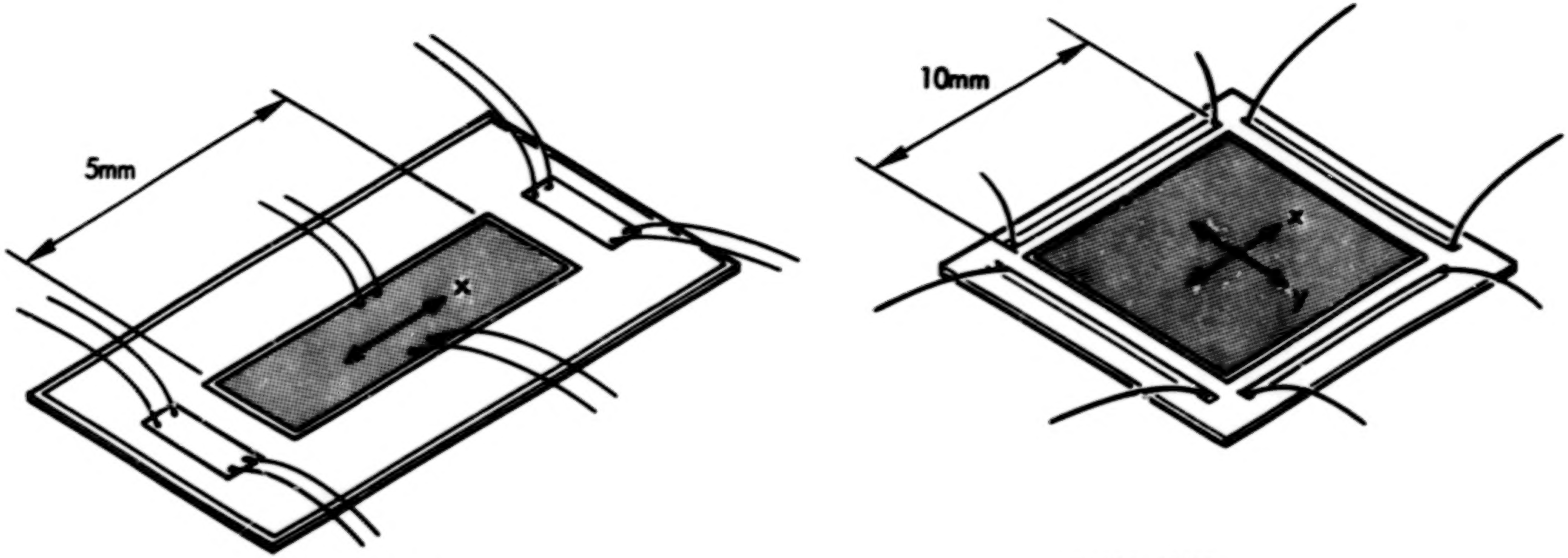
Figure 9

478

Lateral Effect Photodiodes (Figure 10)

The two lateral effect photodiodes illustrated are single-axis United Detector Technology devices. Their principal of operation is quite simple. The substrate between contacts can be assumed to be a uniform resistivity sheet. Each contact feeds a current measuring amplifier and channel electronics. A spot of light imaged in the central sensitive area of the detector injects a current into the resistance sheet, and this current distributes itself at the contacts dependent upon the position of the spot. If, in the single-axis detector, the two contact currents measured are I_1 and I_2, then the position of the spot as indicated by these currents is $X = (I_1-I_2)/(I_1+I_2)$. In the dual-axis device, the X and Y current pairs are treated independently.

LATERAL EFFECT PHOTO DIODES



SINGLE AXIS
PLANAR DIFFUSE SILICON
(UDT-PIN-3244)

DUAL AXIS
PLANAR DIFFUSE SILICON
(UDT-SC/10D)

Figure 10

479

<u>Single-Axis Coordinate Detector Response</u> (Figure 11)

A typical measured response curve is shown for a single-axis device. The response displays a slight sigmoidal bowing about a straight-line best fit.

SINGLE-AXIS COORDINATE DETECTOR RESPONSE

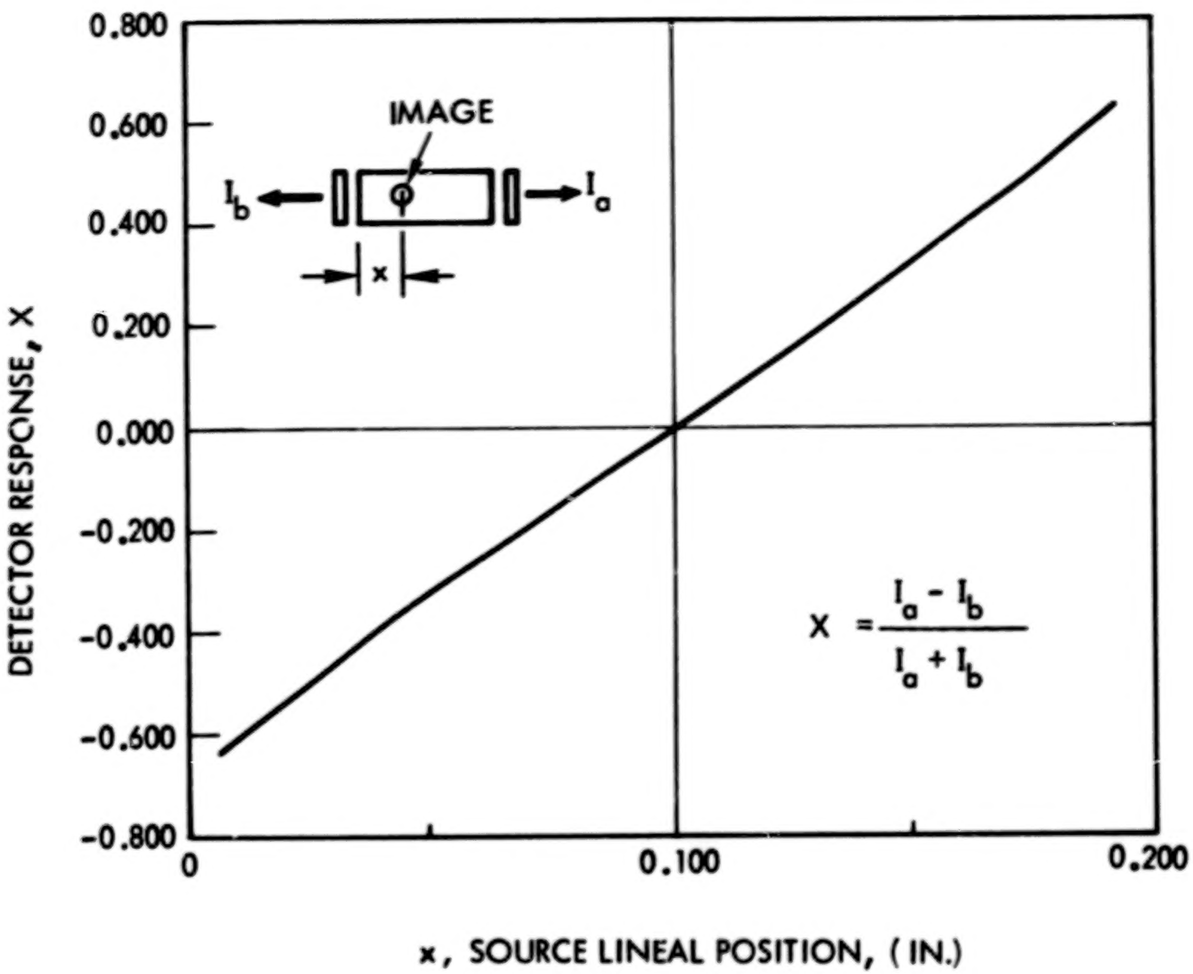


Figure 11

481

Single-Axis Detector Response, Deviation From Linear (Figure 12)

The amplified deviation of the response from a best linear fit is shown in this plot. Neglecting the edge region, the deviation is about $\pm 4 \times 10^{-3}$ referencing full scale. The central portion, however, is quite linear, and suggests that the spot travel be limited to the central .06 to .14 inches.

SINGLE-AXIS DETECTOR RESPONSE
DEVIATION FROM LINEAR

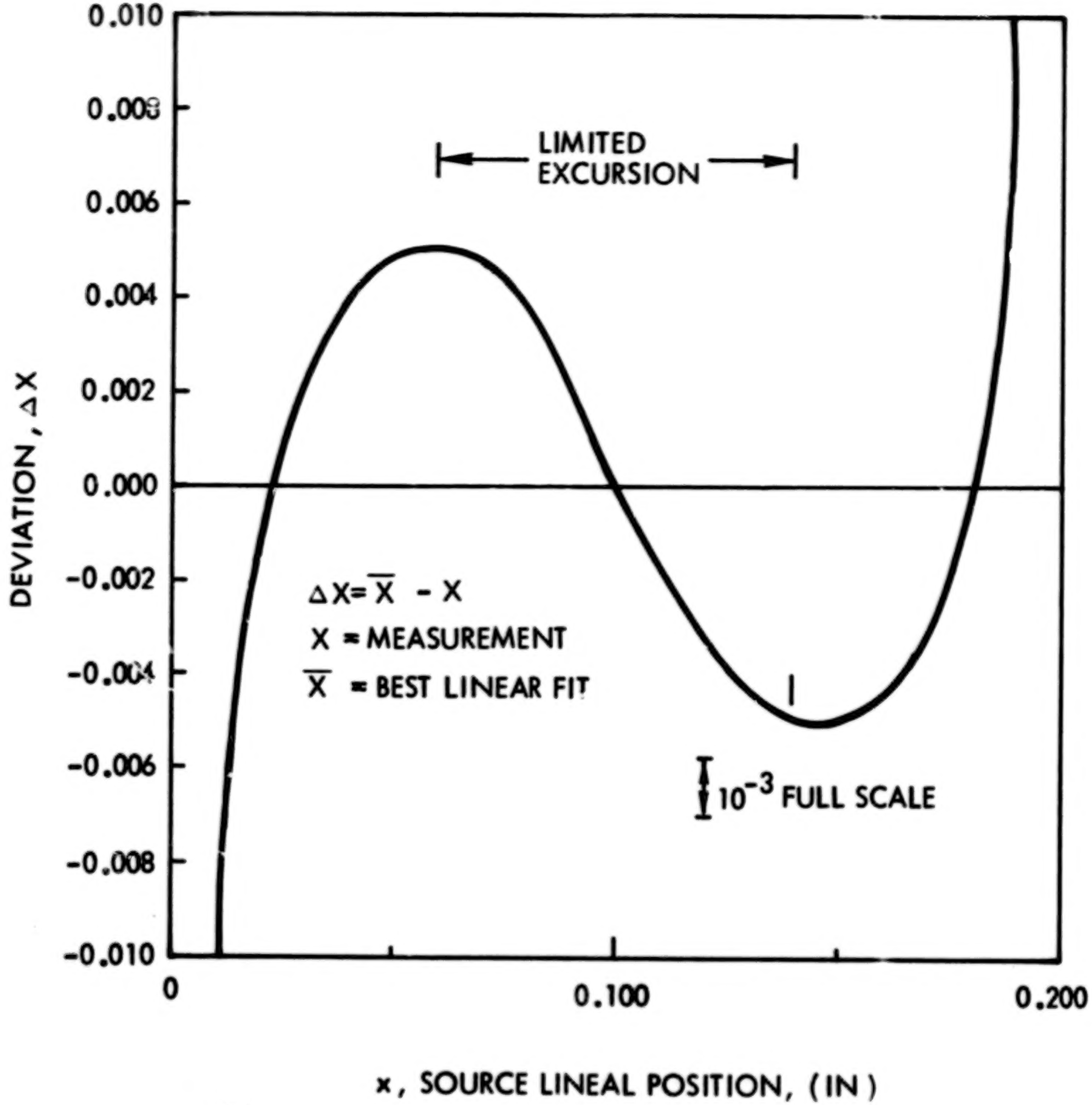


Figure 12

Single-Axis Detector Response

Deviation From Linear With Limited Excursion (Figure 13)

With the limited excursion, the response shows a deviation of about 10^{-3} referencing the restricted excursion. Furthermore, the small deviation of the experimental points from a smooth cubic curve show that such a correction can improve the accuracy to about 10^{-4}. Further tests and modelling may reveal that additional improvement is available with more optimized configurations.

SINGLE-AXIS DETECTOR RESPONSE
DEVIATION FROM LINEAR WITH LIMITED EXCURSION

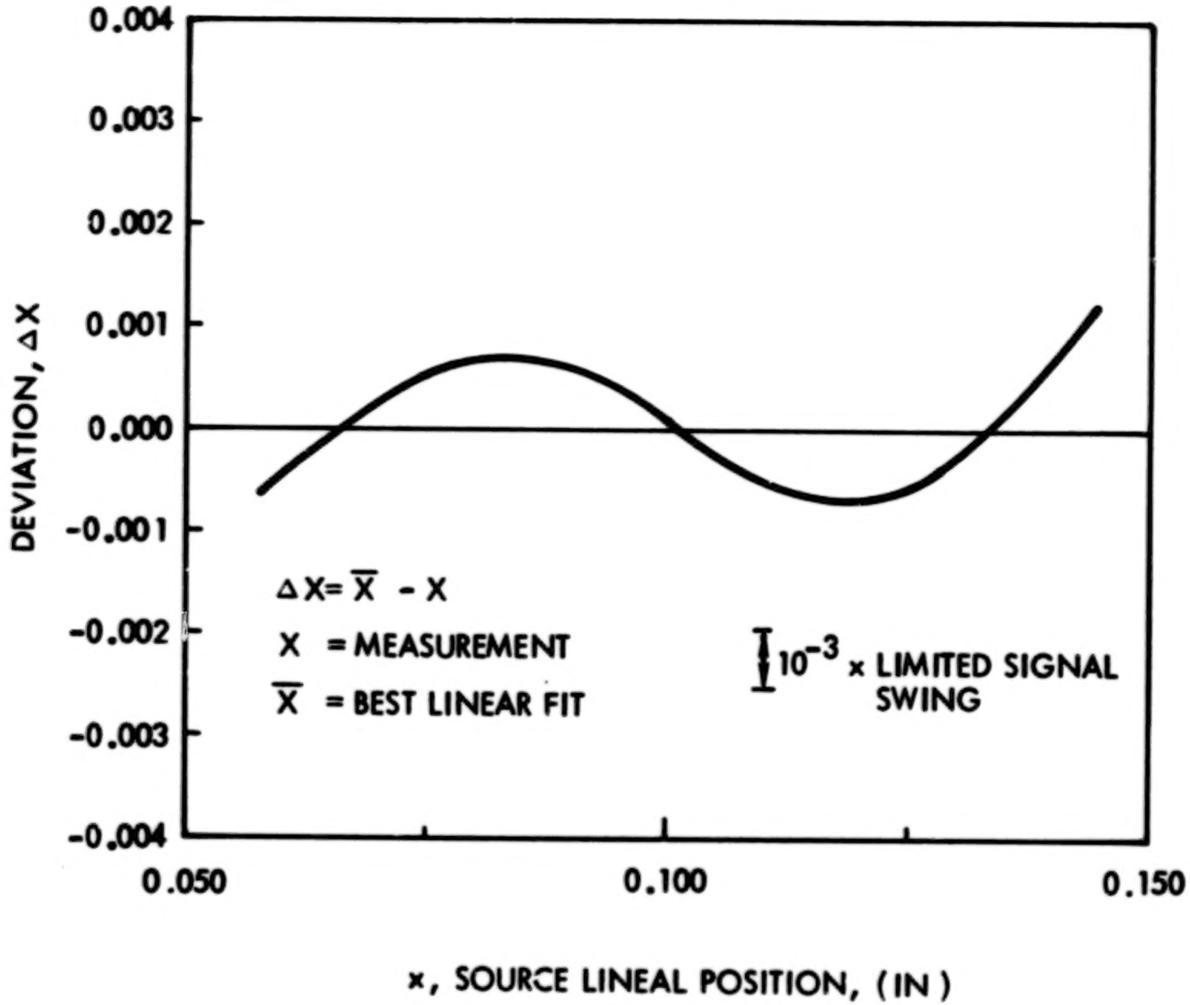


Figure 13

Preliminary Conclusions (Figure 14)

Results to date from tests and modelling the lateral effect silicon photodiode are encouraging. As the conclusions listed show, detector performance and compatibility with other systems requirements appear to meet the demands for many applications; but the detector is only part, albeit an important part, of the sensor system. In addition to further investigations in modelling and testing these detectors, optimization of target sources compatible with various applications is needed. Analog and digital channel electronics to accommodate dynamic ranges of 10^4 (and ultimately 10^5) are required. For applications having a multiplicity of targets, compact, simple multi-detector receivers need to be studied and evaluated.

PRELIMINARY CONCLUSIONS

- A MEASUREMENT SYSTEM USING SILICON LATERAL EFFECT PHOTODIODES AT THE RECEIVER AND LED TARGETS CAN PROVIDE REAL-TIME DEFORMATION MEASUREMENTS.

- THESE PHOTODIODES USED IN A SINGLE AXIS DETECTING MODE ARE APPRECIABLY MORE LINEAR (AND PROBABLY MORE ACCURATE) THAN IN A DUAL AXIS MODE.

- SINGLE AXIS MODE TESTS SHOW THE FRACTIONAL ACCURACY (ERROR/MAX. EXCURSION) IS ONE PART IN 10^3 UNCORRECTED, AND ONE PART IN 10^4 WITH SIMPLE CUBIC CORRECTION.

- SINGLE AXIS DETECTOR MODELLING INDICATES THAT AN ACCURACY OF ONE PART IN 10^4 UNCORRECTED AND ONE PART IN 10^5 WITH SIMPLE CUBIC CORRECTION MAY BE AVAILABLE WITH APPROPRIATE DETECTOR CONFIGURATION.

487

Figure 14

PRELIMINARY CONCLUSIONS

- REJECTION OF SUNLIT GLINTS, EARTHSHINE, AND OTHER NATURAL BACKGROUND CAN BE EFFECTED BY MODULATION ENCODING LED TARGET SOURCES.

- EXCLUDING PREAMPLIFICATION, ELECTRONIC PROCESSING IS DIGITAL AND THEREBY COMPATIBLE WITH MICROPROCESSING AND CONVENTIONAL RECORDING.

- DETECTOR AND ELECTRONIC TECHNOLOGY REQUIRED EITHER EXISTS OR REPRESENTS A MODEST EXTENSION OF STATE-OF-THE ART.

In Flight Optical Measurement of Antenna Surfaces

Photodiode Precision Sensor Elements

An application of a photodiode utilizes a 1 millimeter diameter spot to sense motions in the 10^{-6} meter range. The relatively large spot averages out the fine structuring within the diode and results in a more linear output. The typical application utilizes a light source with a long focal length subtending very small angles and near vertical impingement. A soft or springy mounting requires some technique for eliminating local effects. Flight histories to date have not shown evidence of degradations during a four year period. In the application of these diodes, the resistivity generates the working signal; therefore a uniform change in responsiveness or a decrease in the intensity of the light does not change the accuracy of the system.

STRUCTURAL ALIGNMENT SENSOR

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LOCKHEED MISSILES AND SPACE COMPANY INC.

SUNNYVALE, CALIFORNIA

JANUARY, 1978

"The material presented herein was developed in both an I.D. program and a contemporary contractual DOD program, the latter with much finer distance measurement resolution requirements; the need for a multiple wavelength hierarchy derives from these finer resolution requirements. Coarse resolution (~100-200 micron), which is the function of SAS, requires only a single laser mode together with the R.F. phase modulation/demodulation scheme presented."

COMPARATIVE MICHELSON INTERFEROMETERS (Figure 1)

Length is measured by using a modified Michelson interferometer technique. Figure 1 (a) is a "bare bones" sketch of the conventional Michelson. The main elements of the Michelson interferometer are, (1) a beamsplitter/recombiner, (2) a reference path with a mirror which returns the reference beam on itself, (3) a working path with a movable mirror which returns the working beam on itself, and (4) a monochromatic, highly coherent light source oriented to allow the beams to return "on themselves". When the superposed pair of returned beams are either in or out of phase, then the electromagnetic wave amplitudes, respectively, add or subtract and bright or dark fringes, respectively, are the result. Observe that input and output are both in the optical domain. Figure 1 (b) is a basic heterodyne form of Michelson interferometer. Observe that its upper part is identical to Figure 1 (a). However a local oscillator (L.O.) beam, at a displaced frequency, is combined with the output beam of the upper part with the aid of a second recombiner element. This doubly combined beam is concentrated on a photodetector which acts as an ideal square law detector for electromagnetic waves. Its output contains an electronic intermediate frequency (IF) signal at the optical difference frequency. This signal preserves all amplitude and phase relations which exist in the optical domain. The phase angle due to the range length, R, generated on an optical beam of real or synthetic wavelength, λ, has the general form

$$\phi = 4\pi\frac{R}{\lambda}$$

Thus. a precise determination of ϕ is functionally equivalent to a determination of R.

COMPARATIVE MICHELSON INTERFEROMETERS

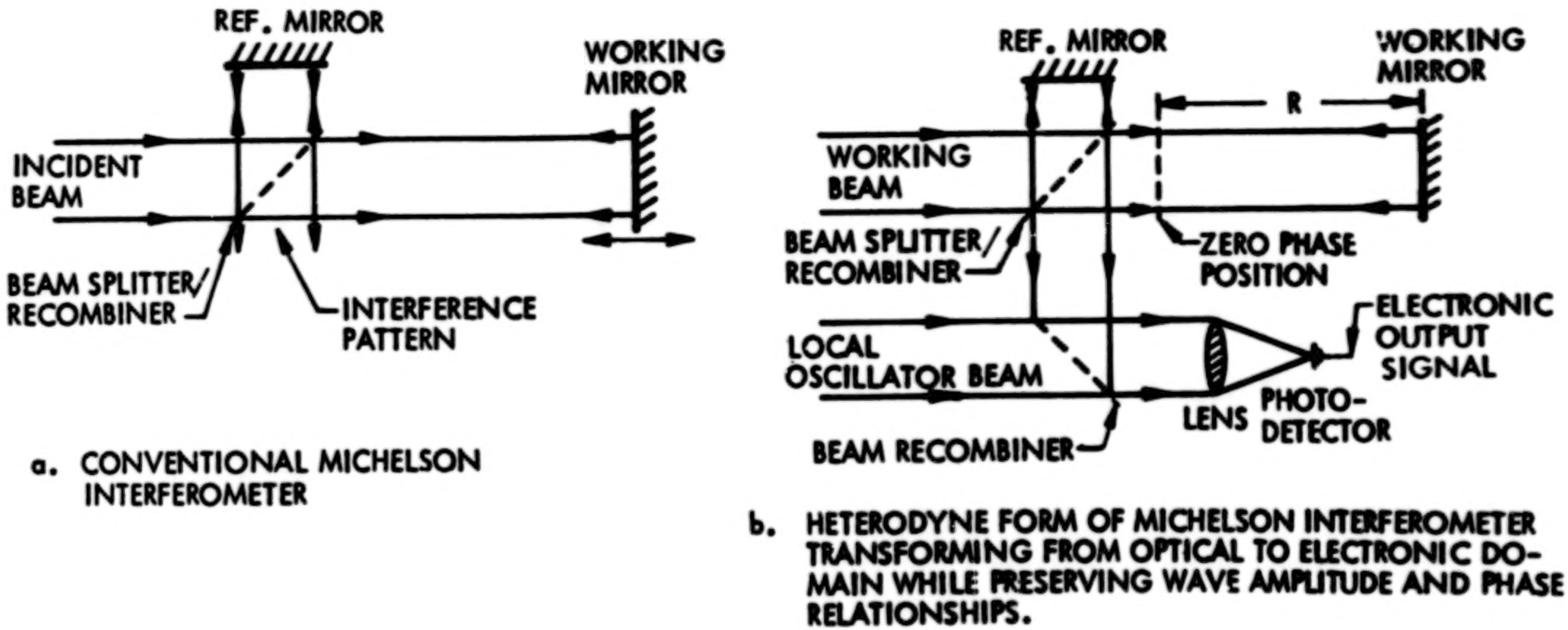

Figure 1

494

SAS OPERATING RANGE POTENTIAL (Figure 2)

The interferometer works over a hierarchy of wavelengths from quite coarse to fine optical wavelengths with some adequate number of intermediate wavelengths. These intermediate wavelengths may be synthetic in nature, a term which is defined below. The multiple wavelengths are generated in three distinct ways. Starting with the shortest, these are real wavelengths of a CO_2 laser. In our usage, the transitions R(16), P(18), and P(20) can all be generated. What is believed unique is that two wavelengths with common polarization can be excited simultaneously, and our laser can be switched from one mode pair to another at a fairly rapid switching rate.

The second way of generating wavelengths is by interaction of the real wavelengths above and generation of beat frequencies. An optical beat or difference frequency leads to an effective or synthetic wavelength, λ_{eff}, of the form

$$\lambda_{eff} = \frac{\lambda_1 \quad \lambda_2}{\lambda_2 - \lambda_1}$$

where λ_1 and λ_2 are the pair of real laser output wavelengths. The R(16), P(18), and P(20) wavelengths are, respectively, 10.28, 10.57, and 10.59 microns. Synthetic wavelengths corresponding to specific wavelength pairs are 343.8 microns for R(16)-P(18) and 5.6 mm for P(18)-P(20). These synthetic wavelengths are, respectively, 1-1/2 and nearly 3 orders larger than the optical wavelengths. They constitute the intermediate wavelength set mentioned earlier.

The third way of generating wavelengths is by the phase modulation of a laser frequency. A 100-MHz sinusoidal modulation generates an equivalent wavelength of 3.0 meters with a fringe length of 1.5 meters. Observe that such a wavelength is about 3 orders larger than the P(18)-P(20) synthetic wavelength. Collectively, the total group of wavelengths bracket the length range from macroscopic (gross) to microscopic and potentially submicroscopic (fine) domains. The macroscopic end of the range can be increased to any desired value by decreasing the modulation frequency.

SAS OPERATING RANGE POTENTIAL

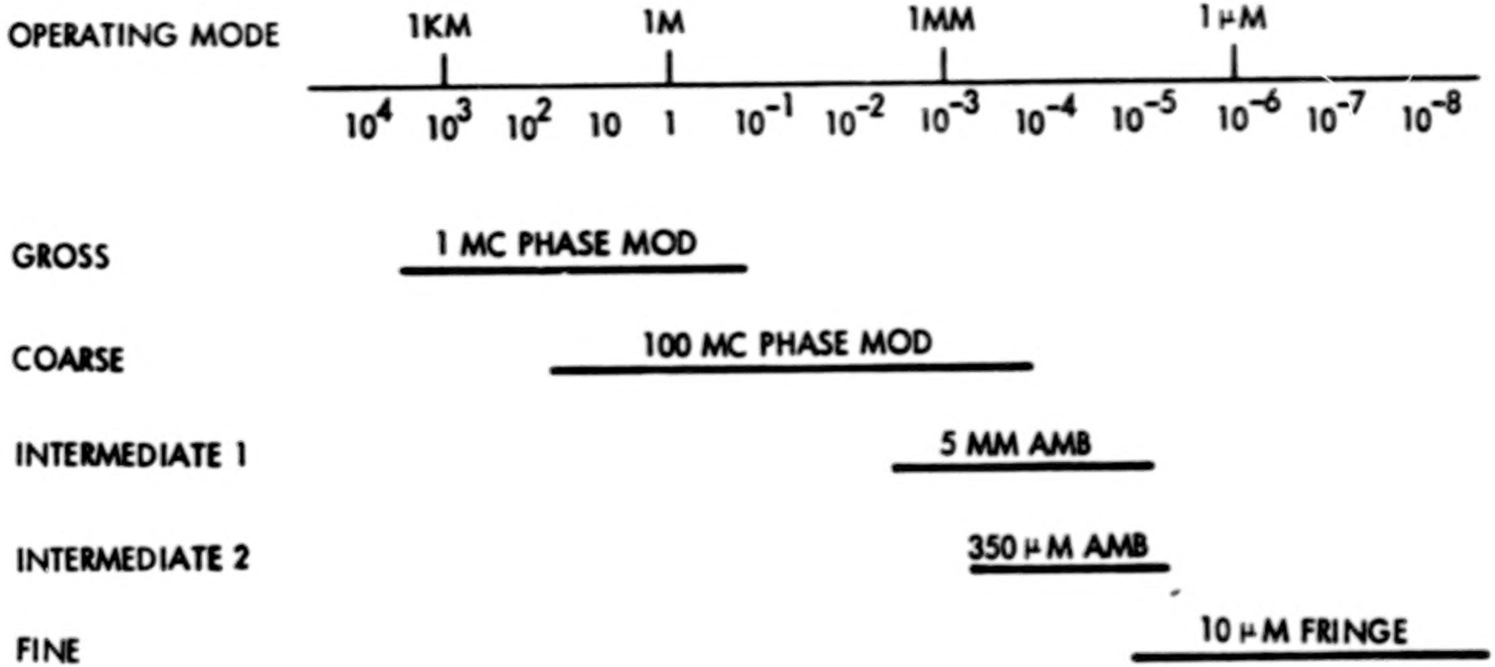

Figure 2

SAS DISTANCE MEASUREMENT LOGIC (Figure 3)

The interferometer works over a hierarchy of wavelengths from gross or coarse to as fine as required. Starting with the coarsest wavelength (first level), one must both count the integral number of fringes (i.e., half-waves) and measure with sufficient accuracy the fractional fringe length. Sufficient accuracy means less than half a fringe of the second level fringe size. As a result, we determine the length with enough accuracy to count, without error, the integral number of second level fringes contained within the length. At this point, the second level fringes take on the role of first level and are used to determine the integral number of third level fringes plus their fractional increment. This procedure is continued until the length measurement with desired accuracy can be made.

This scheme may be compared to a pair of staircases in which we descend in fringe length while ascending in fringe count simultaneously. It is clear that the critical requirement of the scheme is that the measurement accuracy at each hand-off to the next level fringe must be sufficient to ensure no loss of a fringe or stated differently no error in the integral fringe count. This requirement imposes derivative requirements on the accuracy with which each wavelength is known and on the measurement accuracy.

DISTANCE MEASUREMENT LOGIC SCHEMATIC

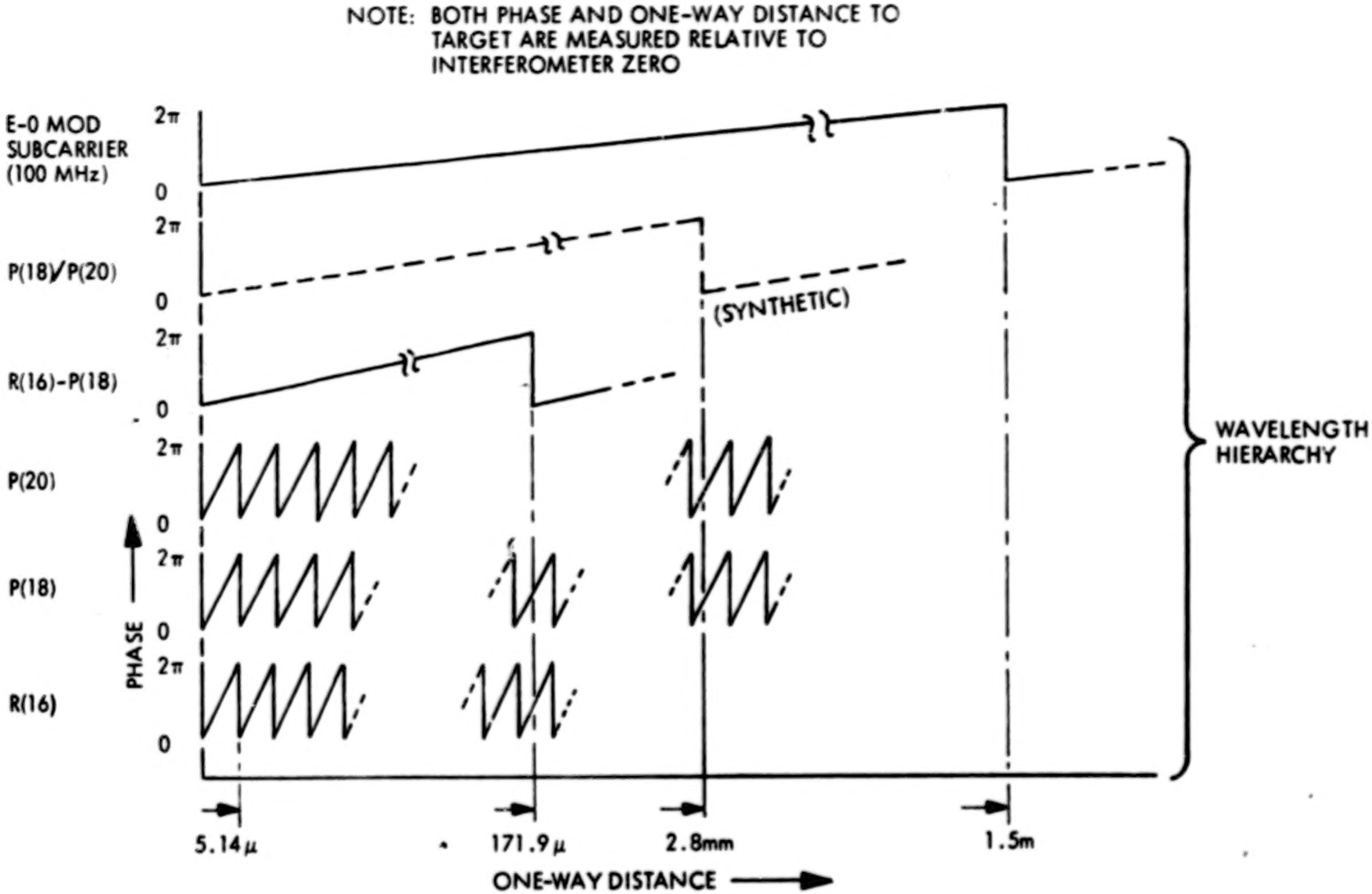

Figure 3

497

498

THE BASIC SAS SYSTEM (**Figure 4**)

This consists of a modified heterodyne form of Michelson interferometer which is capable of performing precise measurement of distance to any one target site in all-electronic fashion and with no moving parts. It uses a two-color laser source suitably isolated from perturbing return radiation and R.F. modulated to provide a macroscopic synthetic wavelength. A movable pupil, not shown, or an alternate method selects a small cross section of the beam for use in the interferometer configuration. By the choice of beam cross section position, the working beam is aimed at the desired target site where a small retroreflector may be positioned. Part of the returned reference and working beams are combined by the beamsplitter, and sent on to the photodetector for heterodyne detection and subsequent signal processing, distance determination, and possible control functions.

THE BASIC SAS SYSTEM

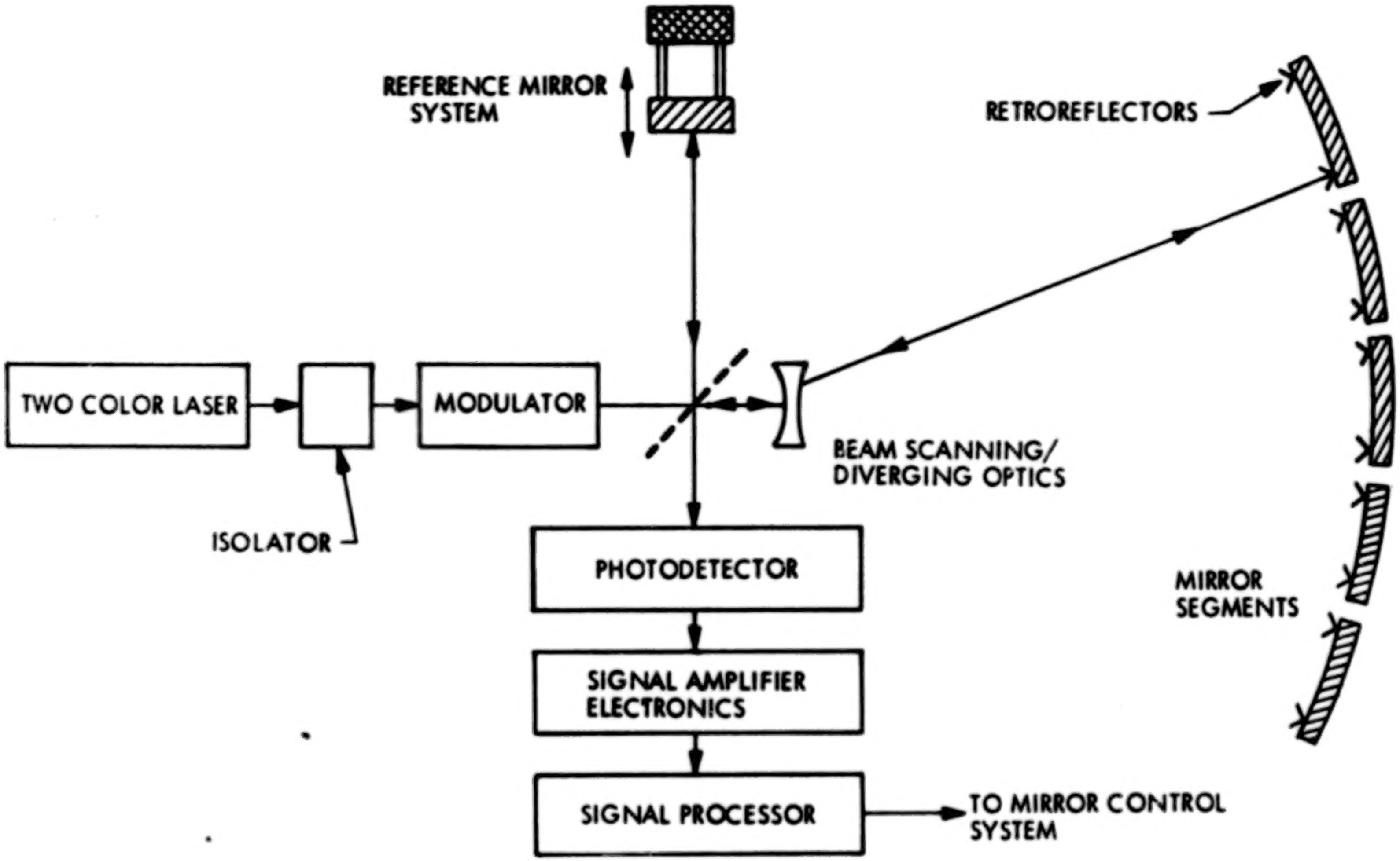

Figure 4

499

500

SAS OPTICAL LAYOUT DIAGRAM (Figure 5)

The optical configuration shown is formed of these functional blocks: A switchable, two-color laser light source; Bragg cells providing frequency translation for the working beam with protective isolation of the source and a coarse wavelength generating phase modulator; beam expanders and field scanning pupil apertures to illuminate the desired target; the modified Michelson interferometer module; and the heterodyne detector array after which our information is in the electronic signal domain. That portion of the laser beam not frequency translated acts as the local oscillator for subsequent heterodyne detection. The RF phase modulation provides the wavelength for the first level or coarse scale. As shown, the 100-MHz modulation frequency yields a synthetic 3.0 meter wavelength or a 1.5 meter fringe distance. The grating/dual detector configuration provides the wavelengths for fine scales. From simultaneous observation at a pair of optical wavelengths, one detects beat frequencies (and related phases) corresponding to synthetic wavelengths for the intermediate scales.

SAS OPTICAL LAYOUT DIAGRAM

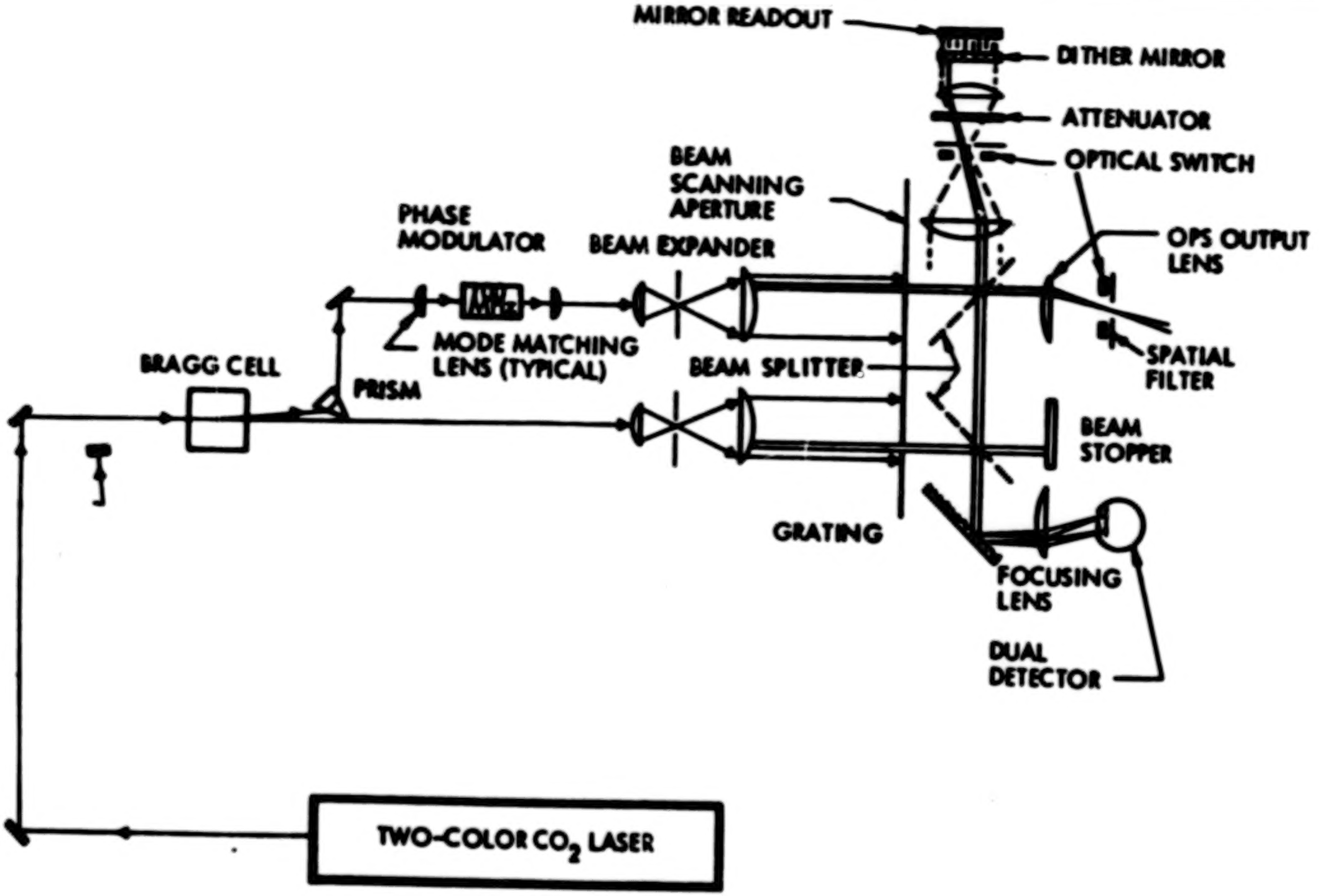

Figure 5

502

COARSE MEASUREMENT SIGNAL PROCESSOR (Figure 6)

The configuration shows the processing flow. Beginning in the upper left-hand corner, modulation frequencies of 1.0 or 100 MHz are selectable by the RF switch. The selected RF power is divided and the first fraction passes successively through an adjustable (phase shifting) transmission line, a power amplifier, and a phase modulator for the working beam. This portion of the circuit provides a sinusoidal phase modulation. The other fraction of the RF power passes to the 90° hybrid divider where about one-half the power is phase shifted and two outputs corresponding to sine and cosine functions are provided. These outputs, each with a phase and amplitude trimmer, go to the inputs of a pair of SPST RF switches. The switch output provides the following RF mixer with sine and cosine inputs on alternate half cycles. Its output after the following bandpass filter corresponds to the functions above and below the bandpass filter block. The crystal detector, amplifier, and logarithmic voltmeter output, when exhibited on an oscilloscope, provides a signal simular to the upper right-hand inset graph.

Using the adjustable lines, phase shifting is used to move the observation close to a phase angle of $\frac{\pi}{4}$ or $\frac{\pi}{4} \pm N \times \pi$. The phase shifting can be accomplished by several methods and the switching of calibrated delay lines is a case in point. Whatever technique is used requires that the phase shift calibration and stability be compatible with the range resolution required. For the case of switched delay lines used as an example, this applies to both delay lines and the switches. The goal of this phase shifting is to leave a residual phase angle to be measured whose tangent squared function is close to unity and whose step size on the inset figure is close to zero. This permits linearization of the tangent function and avoids the need for using trigonometric functions in automatic data processing associated with the phase measurement and length calculation.

COARSE MEASUREMENT SIGNAL PROCESSOR

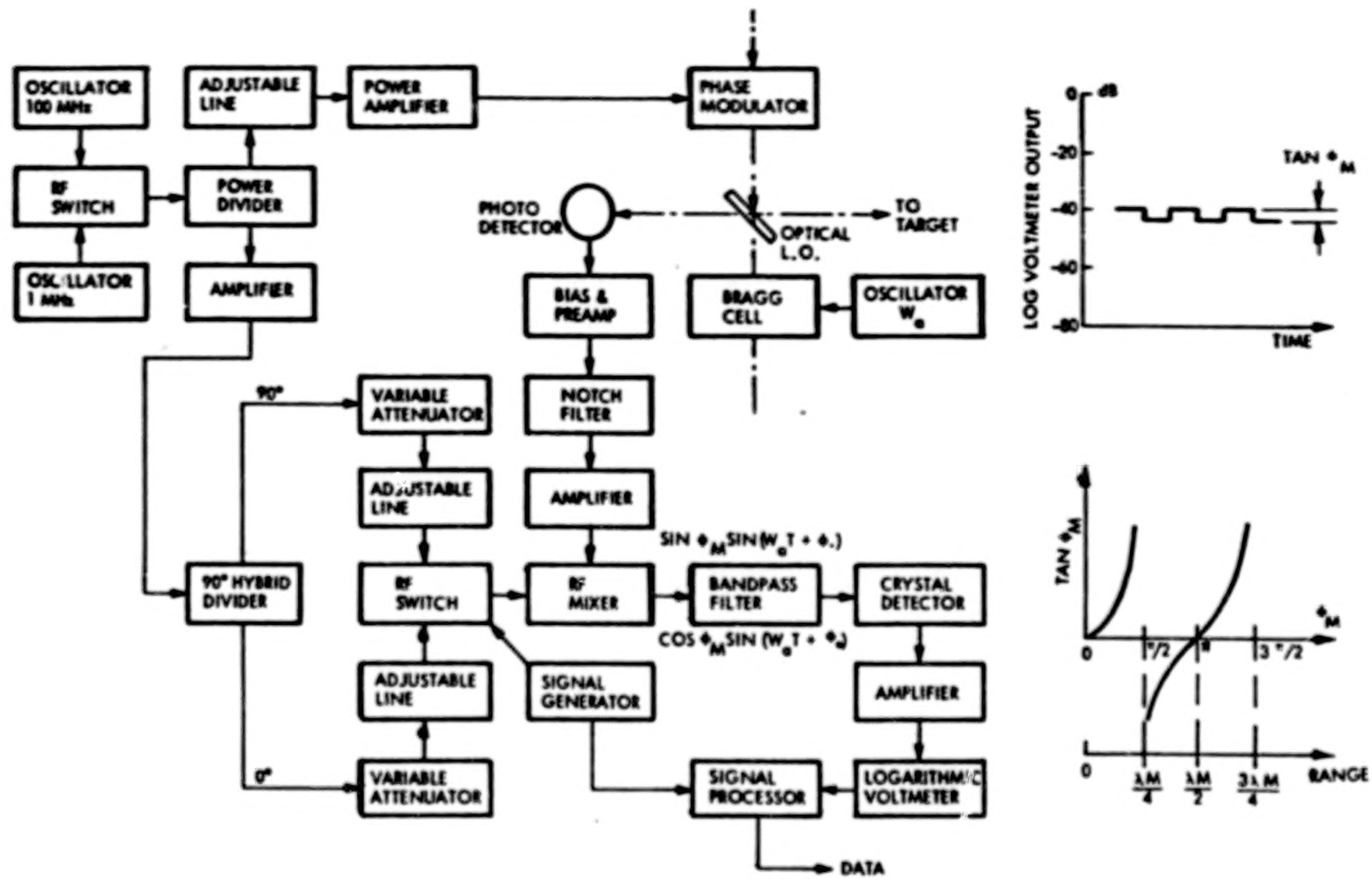

Figure 6

MEASURED RANGE RESOLUTION (Figure 7)

Measurements with laboratory breadboard equipment implementing the sensor concept have been made. The figure represents an experimental oscillographic observation of an output like the inset of the last figure. The upper trace represents the case where phase angle is adjusted to $\frac{\pi}{4}$ or a zero height step in the display. For the lower trace, the target range was displaced by 1.7 mm, relative to the previous position. Comparison of the observed step size with the rms noise shows the range resolution to be less than 200 microns. These observations used a data rate of 100 readings/second and an electronic bandwidth of 1.0 Hz. The resolution obtained is typical of the current capability of the equipment and within a factor of two of the resolution limit estimated in the theory developed for this sensor. Noise smoothing efforts, currently underway, are expected to improve on this resolution.

MEASURED RANGE RESOLUTION

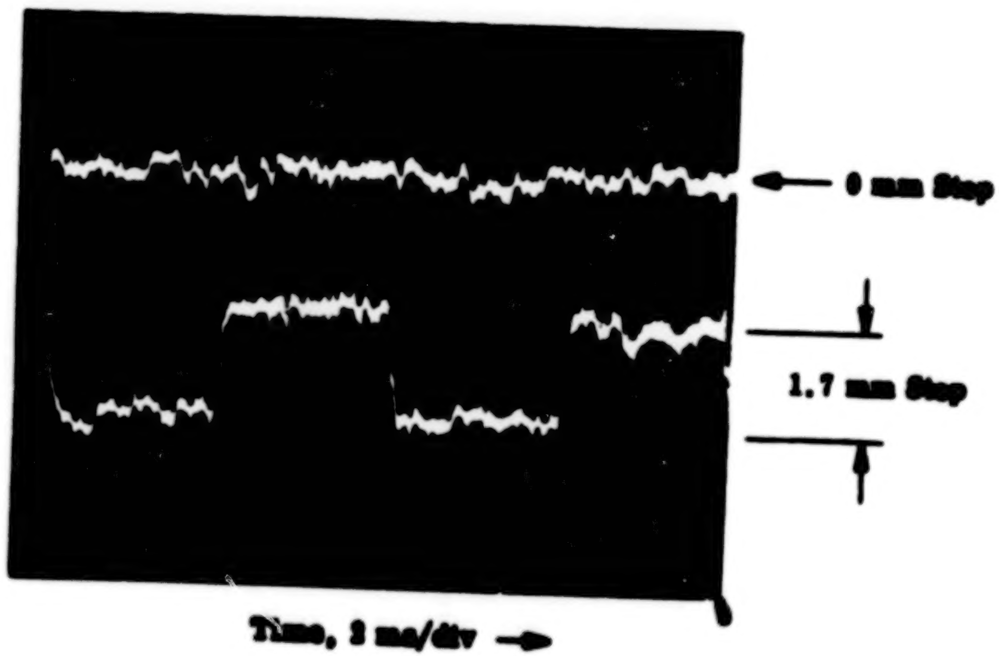

Figure 7

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506

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FUTURE LARGE SPACE SYSTEMS OPPORTUNITIES -

A CASE FOR SPACE-TO-SPACE POWER?

PRESENTED AT THE

GOVERNMENT/INDUSTRY SEMINAR

ON

LARGE SPACE SYSTEMS TECHNOLOGY

AT

LANGLEY RESEARCH CENTER
JANUARY 17-19, 1978

L. B. GARRETT AND W. R. HOOK
LANGLEY RESEARCH CENTER

507

508

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INTRODUCTION

The requirements for space power and energy are forecast to increase exponentially over the next few decades. In this presentation we examine applications and options for beaming power to near-Earth space users from a central space power platform. The cost effectiveness of on-board versus remote power transfer is examined for orbital transfer propulsion systems. Performance characteristics are projected for advanced power generation, transmission, and receiver systems for the 1990's. Major technological development needs are identified with particular emphasis on large space systems technology.

Historically, large space structures technology has been driven more by energy and power systems than by any other spacecraft system (e.g., Skylab's solar arrays including the Apollo telescope mount arrays were over 200 square meters in area). This trend is likely to continue in the near term should the Agency's plans materialize for the 25 kw power module to augment shuttle orbiter power and for the 250 kw power system to support space construction base activities.

A system the size of the 250 kw power system could provide the LSST program with significant near-term technological challenges. Over the longer term, as we move into significantly higher power levels and consider the prospects for beaming power over long distances in space, the technology development needs will, without question, increase by orders of magnitude.

FUTURE NEAR-EARTH SPACE ENERGY NEEDS (Figure 1)

Two classes of future energy users are shown in this figure; the lower region corresponding to non-propulsive orbiting electrical users and the upper region corresponding to propulsion missions such as orbit-to-orbit transfer, comet rendezvous missions, and possibly lunar, solar and interplanetary missions. (No programmatic implications are intended in this projection.) The trend is toward exponentially increasing space energy requirements reflecting higher power, longer duration missions, and larger, heavier payload opportunities provided by the Space Transportation System. We also note that energy requirements for accelerating spacecraft are typically higher than energy needs of nonpropulsive electrical users and are expected to remain so in the future. In 1990, the lower boundary of the orbit-to-orbit propulsion energy requirements curve corresponds to the amount of electrical energy required by a Solar Electric Propulsion System (SEPS) orbital transfer vehicle to move 10^5 kg (the equivalent of two Apollo spacecraft) annually from Low Earth Orbit (LEO) to Geosynchronous Earth Orbit (GEO) and return. The upper boundary at the year 2000 corresponds to the energy required to move 10^8 kg (one Satellite Power Satellite (SPS) using microwave transmission of energy to Earth for terrestrial users) to GEO.

Accelerating spacecraft should make the best case for space-to-space power transfer. This is the application we will examine in this presentation in comparing the relative cost competetiveness of remote versus on-board power. The scenerio assumes the movement of 10^6 kg of payload annually from LEO to GEO beginning in 1990 and increasing at a ten percent rate thereafter. The Orbit Transfer Vehicles (OTV's) to be considered will be an advance chemical OTV, a SEPS, and a remotely powered electrical OTV.

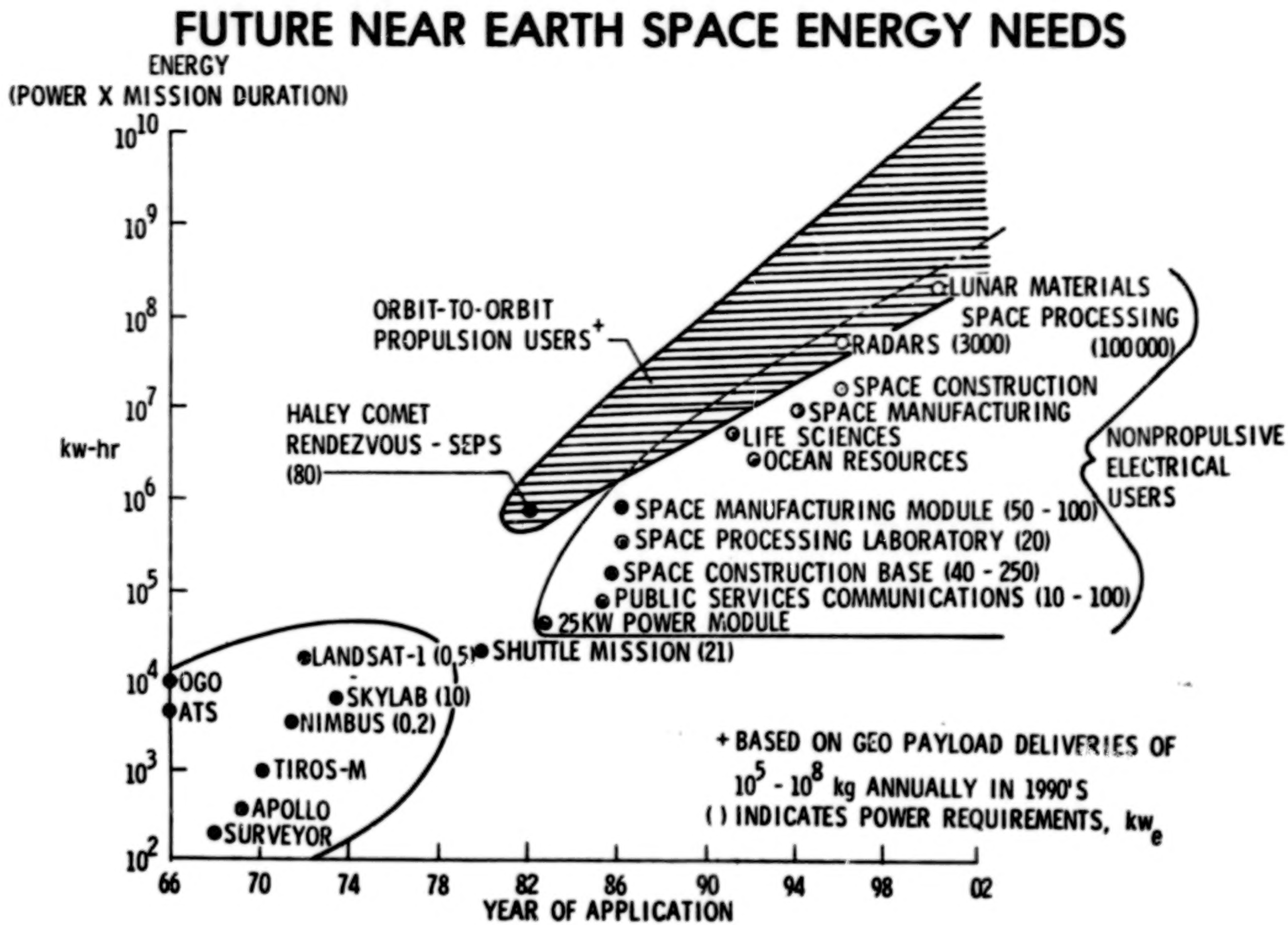

FUTURE NEAR EARTH SPACE ENERGY NEEDS
ENERGY
(POWER X MISSION DURATION)
kw-hr
YEAR OF APPLICATION
ORBIT-TO-ORBIT
PROPULSION USERS+
HALEY COMET
RENDEZVOUS - SEPS
(80)
LUNAR MATERIALS
SPACE PROCESSING
(100 000)
RADARS (3000)
SPACE CONSTRUCTION
SPACE MANUFACTURING
LIFE SCIENCES
OCEAN RESOURCES
NONPROPULSIVE
ELECTRICAL
USERS
SPACE MANUFACTURING MODULE (50 - 100)
SPACE PROCESSING LABORATORY (20)
SPACE CONSTRUCTION BASE (40 - 250)
PUBLIC SERVICES COMMUNICATIONS (10 - 100)
25KW POWER MODULE
SHUTTLE MISSION (21)
LANDSAT-1 (0.5)
SKYLAB (10)
NIMBUS (0.2)
OGO
ATS
TIROS-M
APOLLO
SURVEYOR
+ BASED ON GEO PAYLOAD DELIVERIES OF
10^5 - 10^8 kg ANNUALLY IN 1990'S
( ) INDICATES POWER REQUIREMENTS, kw_e

Figure 1

511

512

SPACE POWER DEVELOPMENT (Figure 2)

Two power generating systems are appropriate at the higher power levels of interest (order of 10's of MW_e to the combined users). They are either based on solar energy collection and generating systems (photovoltaic or solar concentrators providing thermal energy to drive turbogenerators) or a nuclear reactor with its associated power generating system.

The forecast is for incremental increases in installed photovoltaic power to support the nonpropulsive spacecraft electrical needs identified in the preceding figure. Nuclear reactors are expected to be appropriate for applications requiring power above one megawatt. Future reactor improvements are projected to lead to a lower reactor power threshold in the future.

For this study, we will compare two systems - a photovoltaic and a gaseous or plasma core nuclear reactor coupled with a high temperature, high efficiency MHD power generator. In selecting these two systems, both of which have attributes that are appropriate for space applications, we are afforded an additional opportunity to explore the projected performance and relative costs of an existing, incrementally developing technology (the photovoltaic system) with a laboratory or emerging technology (gas core reactor and the MHD generator).

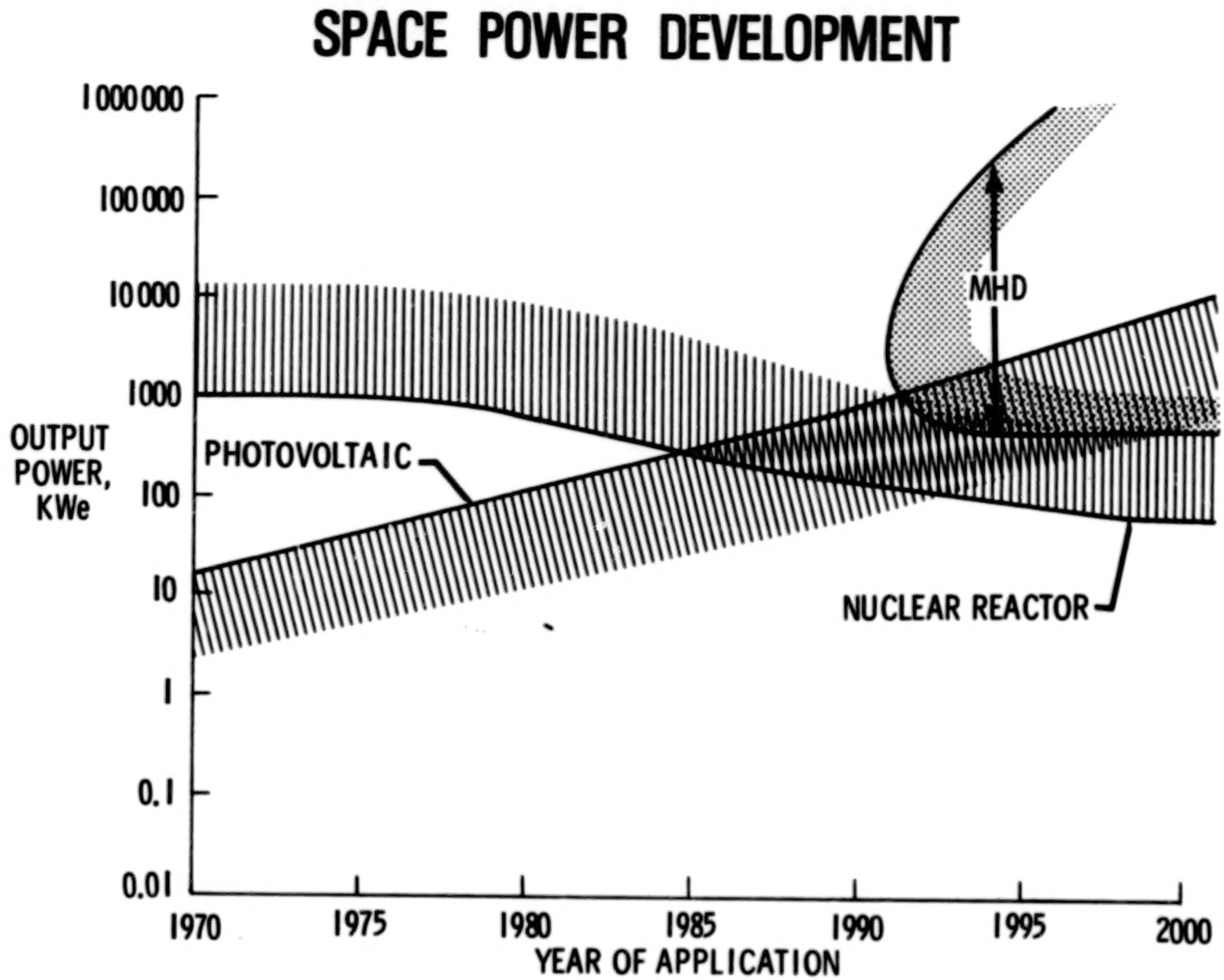

SPACE POWER DEVELOPMENT
1000000
100000
10000
1000
100
10
1
0.1
0.01
OUTPUT POWER, KWe
PHOTOVOLTAIC
MHD
NUCLEAR REACTOR
1970
1975
1980
1985
1990
1995
2000
YEAR OF APPLICATION

Figure 2

513

514

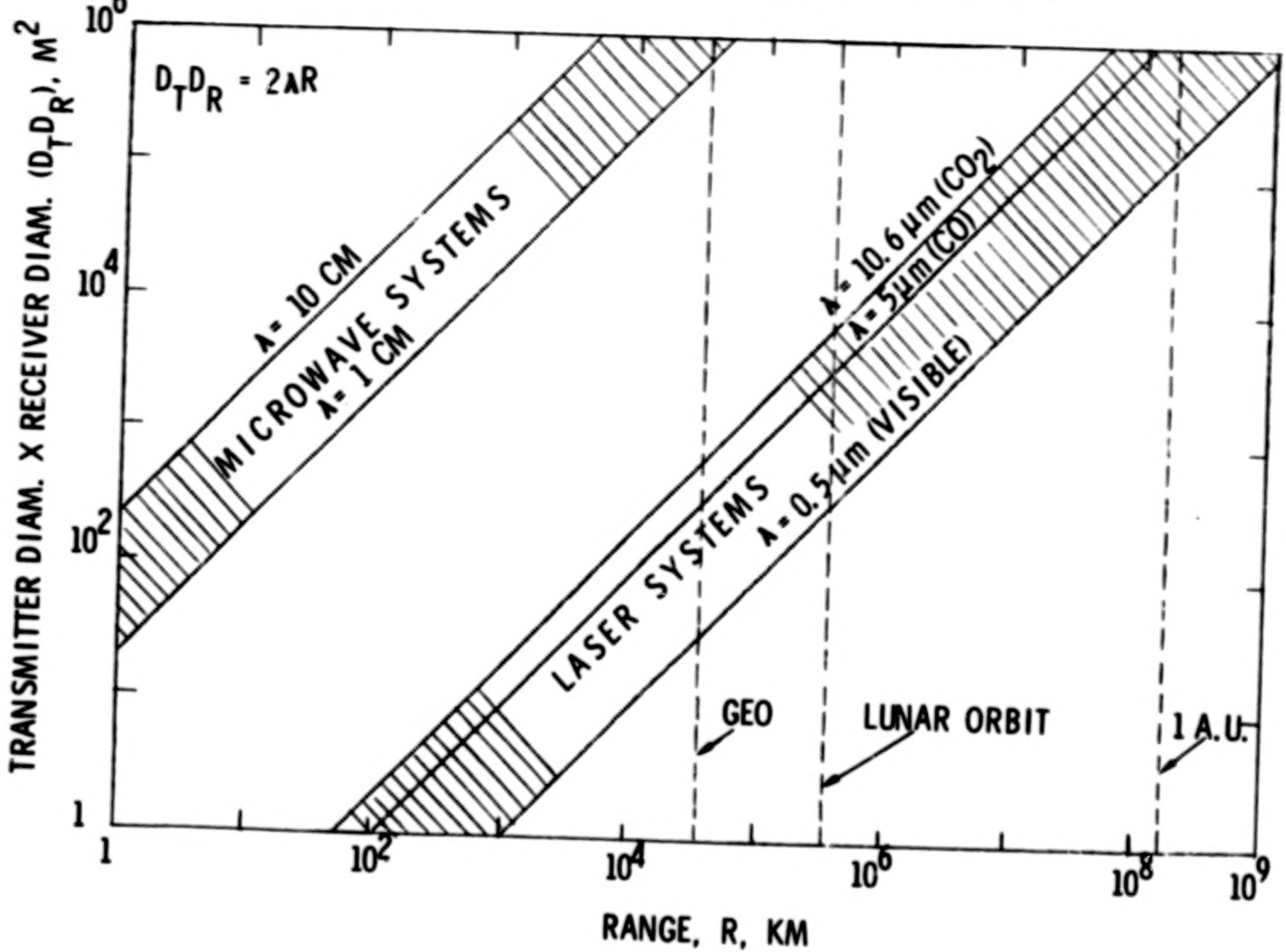
TRANSMITTER/RECEIVER SIZES VS RANGE (Figure 3)

Microwaves or lasers are potential future systems for transmitting power over long distances in space. The size of the transmitter and receiver for such systems are a function of their operating wavelength and the transmission distance or range, not the power level.

To transfer power over geosynchronous distances greater than 40,000 kilometers, microwave transmitter and receiver diameters of 1 to 10 kilometers would be required, whereas laser transmitter and receiver diameters would range from 5 to 30 meters. Nominal transmitter/receiver sizes are projected to be about 2 kilometers and 10 meters in diameter for the microwave and laser systems, respectively. Phased array transmitters are required for the large microwave system and probably will be necessary for the smaller laser system to maintain beam quality.

Consider now the prospects of remote versus on-board power for these two types of receiver systems. For a microwave receiver of 2 km diameter, the equivalent area of on-board photovoltaic cells would produce about 1 GWe of power. In addition, the on-board cells would weigh considerably less than the microwave receiver (by at least one order of magnitude) and would also be less expensive. For the 10 meter diameter laser receiver, the equivalent area for on-board photovoltaic cells would roughly produce about 100 KWe of power. Multiple users in the 100 KWe power range are projected for the future. However, no users requiring 1 GWe power (commensurate with the microwave receiver size) were identified in this study.

TRA"SMITTER/RECEIVER SIZES VS RANGE



515

Figure 3

SYSTEM EFFICIENCIES (Figure 4)

A matrix of projected efficiencies for the photovoltaic and MHD nuclear power generators coupled with the microwave or laser transmission system and the appropriate collection system is shown here. Projected efficiences for the gaseous core nuclear reactor operating at the elevated temperature levels necessary for MHD operation (3,000-4,000 K) could approach 65 percent[1] as compared with 20 percent for advanced solar cells. A laser operating near the visible wavelength region (5,000-9,000 Å) would increase photovoltaic reconversion efficiencies to 40 or 50 percent[2] whereas solar photovoltaic systems which receive energy over the entire solar spectrum can only convert about 20 percent of this incident radiation into useable electrical energy. The bulk of the solar radiation from the infrared spectrum of the Sun goes into waste heat, not electrical energy.

Conversion to microwave energy, transmission and reconversion back to useable electrical energy at the remote user location is projected to be much higher than for the laser system. However, remember that the sizes of the microwave transmitter/receiver will manifest themselves in enormous masses and thus high shuttle launch costs.

---

[1] Williams, J. Richard, et. al.: Satellite Nuclear Power Station: An Engineering Analysis, Georgia Institute of Technology, March 1973.

[2] Stirn, Richard J.: Photovoltaic Conversion of Laser Energy. Second NASA Conference on Laser Energy Conversion. NASA SP-395, 1975, pp. 39-48.

ADVANCED SPACE TO SPACE POWER SYSTEMS

SYSTEM EFFICIENCIES

POWER GENERATION		TRANSMISSION SYSTEM		COLLECTION SYSTEM		OVERALL SYSTEM
PHOTO.	MHD NUC.	MICROWAVE	LASER	RECTENNA	PHOTO.	
20 %		65%		90%		12%
	65	65		90		38
20			30		50	3
	65		30		50	10

Figure 4

CONTENTS

VOLUME I

518

ORBITAL TRANSFER SCENARIO* (Figure 5)

The scenario assumes the delivery of 10^6 kg mass from LEO to GEO beginning in 1990 and increasing at the rate of 10 percent annually thereafter. The payload is delivered in increments of 10^5 kg per trip. Initially, one advanced single-stage orbit-to-orbit chemical OTV is required (assuming a 7-day round-trip time). Alternatively, five solar-powered SEPS or five laser-powered LEPS would be required because of long-trip times (180 days per round trip). For the years 2010 and 2020, payload masses of 6×10^6 and 17×10^6 kg, respectively, would be delivered to GEO. The number of operational vehicles required is shown for each propulsive system. Also noted in parenthesis is the cumulative number of vehicles required for each system. (The chemical OTVs were assumed to have a 50-trip lifetime each, whereas the SEPS and LEPS were good for only three trips each due to Van Allen belt radiation damage to the photovoltaic array.)

Relative cost estimates for on-board versus remote power are developed in the next two figures. A time slice at the year 2010 is used for sizing and costing the competing power plant/transmitter/receiver combinations and for the overall on-board versus remotely powered OTV assessment.

*J. J. Rehder of Langley Research Center, Space Systems Division, Vehicles Analysis Branch, generated the advanced OTV performance data for this scenario.

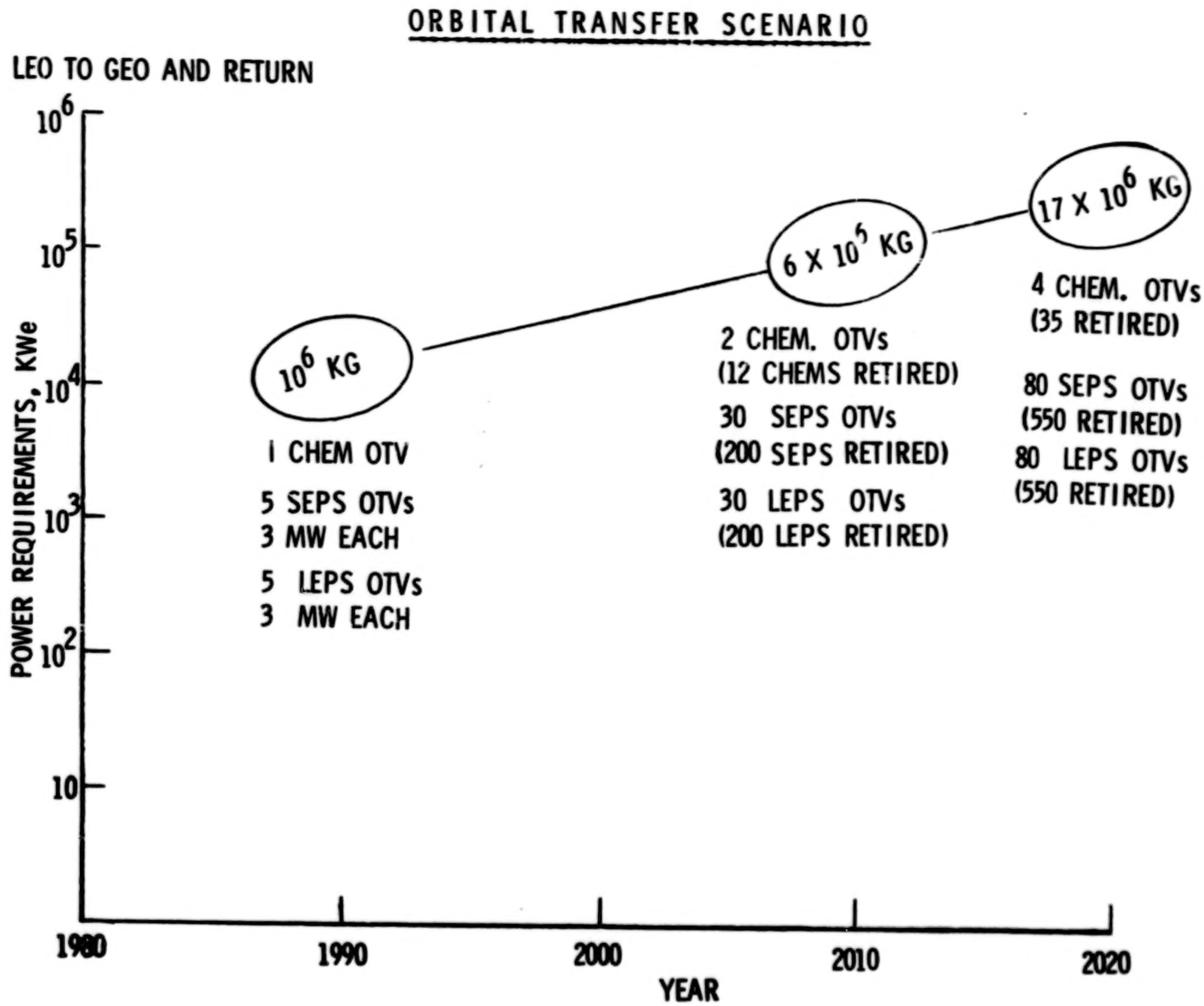

ORBITAL TRANSFER SCENARIO
LEO TO GEO AND RETURN
POWER REQUIREMENTS, KWe
10^6
10^5
10^4
10^3
10^2
10
10^6 KG
1 CHEM OTV
5 SEPS OTVs
3 MW EACH
5 LEPS OTVs
3 MW EACH
6 X 10^5 KG
2 CHEM. OTVs
(12 CHEMS RETIRED)
30 SEPS OTVs
(200 SEPS RETIRED)
30 LEPS OTVs
(200 LEPS RETIRED)
17 X 10^6 KG
4 CHEM. OTVs
(35 RETIRED)
80 SEPS OTVs
(550 RETIRED)
80 LEPS OTVs
(550 RETIRED)
1980
1990
2000
2010
2020
YEAR

Figure 5

POWER PLANT COSTS AND MASS ESTIMATES (Figure 6)

The cost/mass matrix is based on the electrical requirements of the 30 OTV users receiving either microwave or laser transmitted power from the photovoltaic or the MHD gaseous core nuclear reactor space-based power generating systems. Each matrix element contains three entries: The upper numbers are (1) the relative costs of the power plant plus the transmission system including RDT&E costs; (2) the mass of the entire plant and transmission system. The lower number in parentheses is the shuttle launch cost to place the power plant and transmitters in low Earth orbit. Data sources for the costs and mass estimates are referenced below.[1-6]

---

[1]"Outlook for Space;" NASA SP-386, January 1976.

[2]"A Forecast of Space Technology, 1980-2000;" NASA SP-387, January 1976.

[3]"Space-Based Solar Power Conversion and Delivery Systems Study," Volume V, Economic Analysis, ECON, Inc., 77-145-1, May 31, 1977 (NASA Contract NAS8-31308).

[4]"Satellite Power System - Engineering and Economic Analysis Summary." NASA TM X-73344, November 1976.

[5]"Theme Team 7 - Multipurpose Space Power Platform," Report of Presentation to Theme Team Members at NASA/OAST. W. J. Schafer Associates, Inc., WJSA-76-10, May 11, 1976, (NASA Contract NASW-2866).

[6]Williams, J. Richard, et.al.: Satellite Nuclear Power Station: An Engineering Analysis, Georgia Institute of Technology, March 1973.

ADVANCED SPACE-TO-SPACE POWER SYSTEMS

POWER PLANT COSTS AND MASS ESTIMATES

100 MWe TOTAL TO 30 USERS
YEAR 2010

	COSTS*($B)/MASS (Kg X10^6)	
	MICROWAVE TRANSMITTERS	LASERS TRANSMITTERS
PHOTOVOLTAIC GENERATING SYSTEM	\$ 430/ 600 (\$400)+	\$ 1660/ 0.8 (0.5)+
GAS CORE NUCLEAR REACTOR W/ MHD GENERATOR	\$ 17/ 600 (400)+	\$ 5/ 0.2 (0.1)+

★ FIRST UNIT PLUS RDT&E COSTS
+ SHUTTLE LAUNCH COST $B

Figure 6

522

CUMULATIVE SPACE TRANSPORTATION COSTS AT YEAR 2010 (Figure 7)

The OTV costs and the shuttle launch costs to place the 6 X 10^6 kg payload, the OTVs and OTV fuel in LEO are shown. Note that the chemical OTV, because of its long reuse capability, costs only $0.6B compared to $28B for the solar electric OTV or $7B for the laser electric OTV. However, there are large shuttle transportation costs associated with vast quantities of fuel (liquid hydrogen - liquid oxygen) required for the chemical OTV. The overall relative costs, including amortization of the $5B space-based power system, lead to the conclusion that significant cost savings could be realized by beaming laser power from a central utility in space to remote laser electric OTV users. Thirty-year runout costs would yield significantly larger cost advantages for LEPS.

ADVANCED SPACE-TO-SPACE POWER SYSTEMS

CUMULATIVE SPACE TRANSPORTATION COSTS AT YEAR 2010

ORBIT TRANSFER VEHICLE	BILLIONS OF DOLLARS SHUTTLE TRANS. COSTS	OTV COSTS	TOTAL COSTS
CHEMICAL	$ 180	$ 0.6	$180
SOLAR ELECTRIC	61	28.0	89
LASER ELECTRIC	57	7.0	64

Figure 7

LASER ELECTRIC ORBITAL TRANSFER VEHICLE CONCEPT (Figure 8)

A simple concept of a space-to-space power system using laser transmitters for beaming power to laser electric propulsion vehicles is illustrated in this figure. The power system consists of the gaseous core nuclear reactors and MHD generators, a large expanse of thermal radiators (heat pipes), the shadow shield, connecting structure to the habitable module and the laser transmitters. The next level of detail would include the appropriate power distribution subsystem, and on-board controls for the power regulation, monitoring, maintenance, and station keeping functions.

In spite of the high efficiency (65 percent) and high operating temperatures (3,000-4,000 K) of the power generating system, the thermal radiators (200m X 200m for a 600-700 K heat rejection temperature range) are expected to be among the heaviest component in the space power system. Thus, focused technology development and engineering efforts aimed at producing more efficient, lighter weight thermal radiators could yield substantial mass reductions.

LASER ELECTRIC ORBITAL TRANSFER VEHICLE CONCEPT

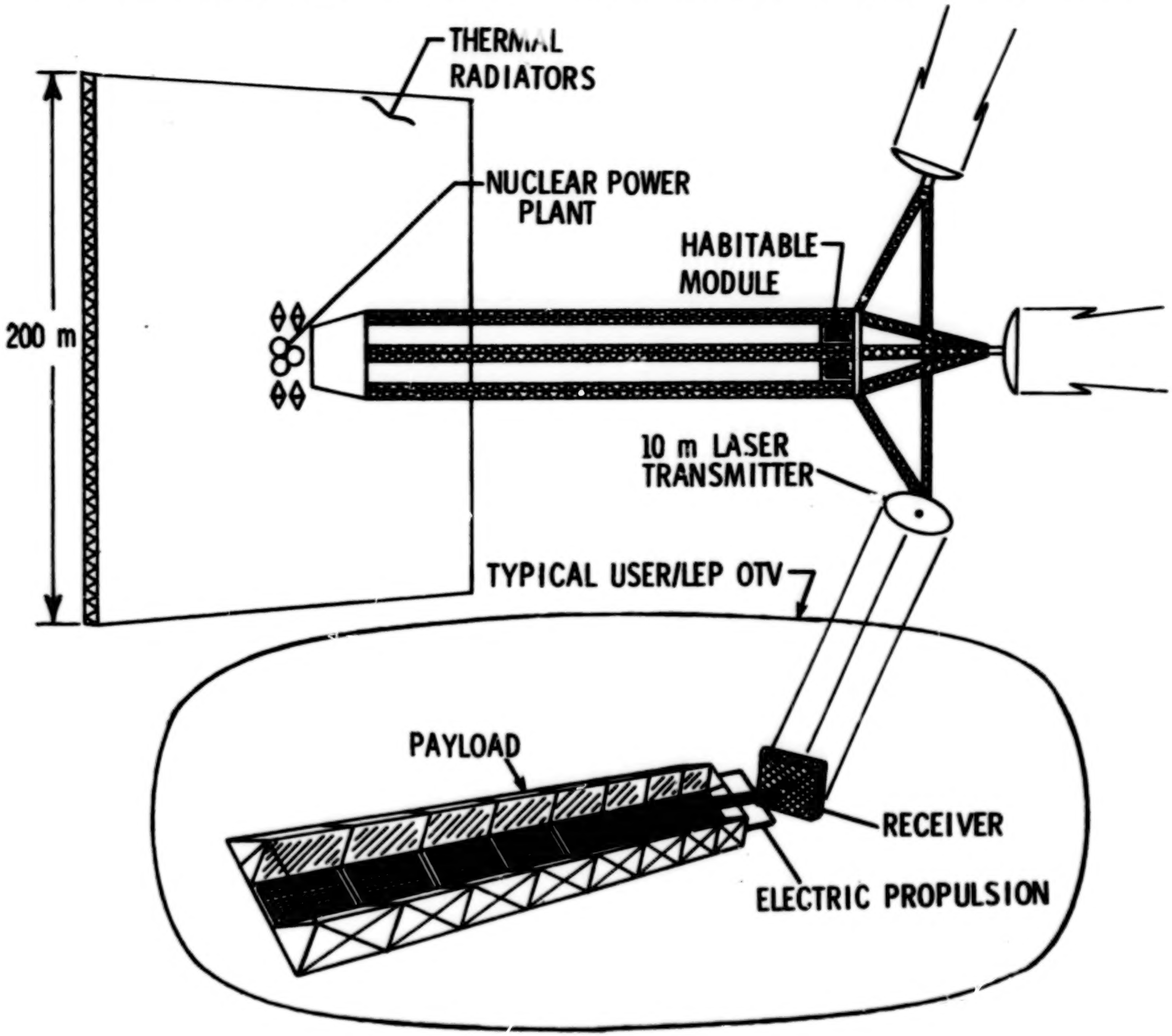


Figure 8

525

526

CONCLUDING REMARKS (Figure 9)

Major technological development needs are summarized for this study on advanced space-to-space power systems. The particular emphasis is on large space systems technology areas.

Study Results and Systems Analysis Needs:

Relative cost estimates were derived for two power generation systems and for two power transmission/receiver systems. Comparable cost estimates were also generated for on-board power and propulsion systems to effect the remote versus on-board power tradeoff.

1. An advanced central space-based power utility, beaming power to remote orbit-to-orbit propulsion users, yields significant cost advantages over on-board power and propulsion systems. For the transportation scenario used in this study, the gas core nuclear reactor and MHD generator with lasers transmitting to remotely located users was shown to be more cost effective than on-board chemical or solar electric propulsion systems for the 1990's. Centrally located photovoltaic power generating systems, even with the projected efficiency improvements and the projected order of magnitude cost reduction for space-rated arrays would suffer severe cost penalties over the nuclear/MHD system or on-board power systems. It was also shown that microwave transmission and collection systems are not attractive for space-to-space power transfer applications. The size and transportation costs of the microwave systems are too great for any future space users examined in this study.

2. It is expected that on-board photovoltaic power generation systems, because of their established record of reliability and attractiveness for numerous space missions, will continue to dominate the high-power areas in the near future. However, cost comparisons indicate that on-board nuclear power systems should be considered as an alternative to large photovoltaic arrays, particularly for the 250-KW power module application and for higher power users.

3. An integrated space transportation system study is required to guide the Agency's future planning and technology development programs. Payload masses could range over several orders of magnitude and the payloads could range from lightweight flimsy structures to heavy concentrated masses. The competitiveness of laser propulsion over chemical and solar electric propulsion vehicles needs to be considered in space transportation studies that extend beyond the year 2000. We also note that the gas core reactor, which in this study was the competitive power generating system, was initially conceived and developed as an efficient thruster for space propulsion. This reactor could play a major role in reducing the future costs of both space power and transportation.

Major Technology Development Requirements:

The gas core nuclear reactors, MHD power generators and laser conversion, transmission and focusing systems are in the laboratory or emerging technology stages. A significant technology development program would be required to produce an operational space-based power system by the 1990's. DOD and DOE R&T programs could support some of the technology development needs. However, the NASA requirements are so dissimilar from those of these two Agencies, it is unlikely that the space-to-space power system will be developed without a substantial NASA effort.

The power, structures, materials and controls subsystems present major technological development challenges to programs such as the LSST. For the most part, the items are self explanatory and consistent with the general requirements for long life, relatively low capital investment and operating costs, and lightweight, closed cycle operation systems.

Significant advances have been made in high power laser development during this decade. Future developments in the 1980's should be even more pronounced. However, much of the recent DOD/DOE emphasis has been on high energy, short duration chemical lasers which are appropriate for military or fusion reaction initiation applications. The NASA requirements for high power, long duration operations would tend to favor the closed cycle operation associated with electric discharge lasers rather than the somewhat more

527

CONCLUDING REMARKS (Continued)

efficient open cycle chemica. [illegible]er systems. (Fuel resupply transportation costs for the chemical laser system would rapidly erode any cost advantages of remote over on-board power systems).

The type of collection system on-board the user spacecraft will dictate the laser transmission spectral frequency or wavelength. In this application, a laser operating in the visible spectrum was selected to produce electrical power to drive ion thrusters directly from a high efficiency photovoltaic receiver. It is also possible to transmit laser energy in the infrared spectrum (CO or CO_2 lasers) to thermal energy receiver systems (such as laser thermal engine or laser thermal propulsion systems). At present, insufficient cost and performance data are available on the laser thermal propulsion system to make reasonable cost tradeoffs for this candidate OTV system. However, the NASA's Lewis and Ames Research Centers are actively pursuing laser thermal concepts. Early study and experimental results for these systems are very encouraging.

The gaseous or plasma core nuclear reactor is one of many types of nuclear reactors which could be cost competitive with photovoltaic systems in the 100's of KWe range and higher. In comparison with solid core reactors, the gas core reactor development is in the embryonic stages. However, it offers several potential advantages over solid core reactors which made it a leading future candidate for long life, high power, space-based power systems. These potential advantages include: (1) compact, power intensive system; (2) small critical mass requirements; (3) automatic fuel reprocessing and breeding for closed cycle operation; (4) eventual extension to ultra high temperature operations for integration with an efficient MHD power generator; and (5) ultimately direct pumping of the laser medium with the nuclear reactor fission fragments (thereby eliminating the electric generation components and the inefficiences and masses associated with these components). Coal fired MHD systems are currently being developed by ERDA for the 1980's. ERDA and NASA are supporting a small effort on gas core reactors and direct laser pumping.

The development of high efficiency, lower cost, radiation resistant solar cells will yield significant benefits in lower cost power to meet future high energy needs of the Agency. In this analysis it was assumed that solar cell damage from the Van Allen radiation belt would limit the SEPS/LEPS OTVs to three trip lifetimes.

Improvements in the radiation resistance characteristics of photovoltaic cells will increase the life expectancy of these OTVs and narrow the cost gap between on-board SEPS and the remotely powered LEPS OTVs. Of course, failure to improve the efficiencies of the solar cells and to reduce the costs of the array by the projected order of magnitude would give the LEPS an even more overwhelming cost advantage over the SEPS.

The development of highly accurate pointing and figure control systems for the laser transmitter may be the most demanding technological challenges for the space-to-space power system. Surface accuracy requirements for the transmitter optics (approximately $\lambda/20$ or 10^{-7} meters RMS) exceed current space system capabilities by several orders of magnitude. Phased array laser transmitters may be necessary to concentrate most of the transmitted energy on the user's receiver system. It is not clear at present whether the space environment will also require active control systems for individual laser mirrors in a phase array transmitter. Receiver surface and pointing tolerances are not nearly as exacting as those for the transmitter and are within current state-of-the-art. The development of automatic feedback and control systems between the receiver and transmitters will be required to keep the laser beams on target.

In summary, the economic advantages and technology and engineering development requirements have been reviewed for the case of power transmission in near-Earth space. A substantial effort will be required to develop the space-to-space power system capability by the 1990's. RDT&E costs are estimated in this study to be $1.5 billion for the gas core reactor/MHD generator and $0.5 billion for the laser generator/transmitter systems. Nonetheless, the results of this study lead us to the conclusion that the long-range cost benefits of 10's to 100's of billions of dollars could justify a focused R&T effort on space-to-space power systems.

SPACE TO SPACE POWER SYSTEMS

CONCLUDING REMARKS

SYSTEMS ANALYSIS

- o ACCELERATING SYSTEMS: BEST CASE FOR SPACE-TO-SPACE POWER
- o 250 KW POWER MODULE NEEDS STUDY
- o NEED INTEGRATED TRANSPORTATION SYSTEM STUDY

POWER

- o LASER: CLOSED CYCLE - LONG LIFE - HIGH POWER
- o NUCLEAR POWER: GAS CORE REACTOR - MHD
- o SOLAR ARRAYS: HIGH EFFICIENCY - LOWER COST - RADIATION RESISTANCE

STRUCTURES

- o LARGE STRUCTURES WITH LARGE MASS CONCENTRATION
- o FIGURE CONTROL MIRROR ARRAYS
- o THERMAL RADIATOR DESIGNS

MATERIALS

- o RADIATION RESISTANT STRUCTURAL MATERIALS
- o HIGH TEMPERATURE MATERIALS MHD AND LASER APPLICATIONS

CONTROL

- o POINTING AND FIGURE CONTROL: LASER TRANSMITTER/RECEIVER
- o POWER AND ENERGY REGULATION/MANAGEMENT

Figure 9

COMMENTS OF GENERAL INTEREST FROM QUESTIONS AND ANSWERS

Future Large Space Systems, Space to Space Power

Alternatives in the Trade Study

This study recognized that concepts for a heavy lift launch vehicle existed and were in work; however, the economics associated with utilization of a heavy lift were not part of these considerations. In a similar manner, the on-going effort toward improved gas reactor technology and improved laser efficiencies make projections which could have influenced this trade study. At this time, such efforts did not have sufficient definition to include in the study. Considerations of other advances such as thermal annealing of solar cells or the use of solar concentrators were recognized as elements which could potentially modify the results of this study.

532

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POST LANDSAT D
ADVANCED CONCEPT EVALUATION

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NASA GODDARD
SPACE FLIGHT CENTER

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GENERAL ELECTRIC,
SPACE DIVISION

533

JANUARY 18, 1978

534

This paper presents some preliminary results of a study being performed by the General Electric Company for NASA's Goddard Space Flight Center under Contract No. NAS 2-9580. Although the PLACE study will discuss technology requirements in a number of areas, this study deals specifically with requirements in the area of large space structures.

(Figure 1)

The principal objective of the PLACE study is the forecasting and identification of the most important technology requirements of this era. Since the critical technology areas are to be based on potential earth resources satellite systems, a second objective of this study will be the creation of a plausible scenario of these future system opportunities, referred to as a "Space systems technology model".

PLACE STUDY OBJECTIVES

- TO FORECAST AND IDENTIFY THE KEY TECHNOLOGIES OF EARTH RESOURCES SATELLITE SYSTEMS OF THE POST '84 PERIOD (1985-2000)
- TO PROJECT COMPREHENSIVE 'SPACE SYSTEMS TECHNOLOGY MODELS' FOR EARTH RESOURCES PROGRAMS FOR THIS PERIOD

Figure 1

536

(Figure 2)

This figure depicts the progression or flow of the PLACE study. Groundwork for the system concepts was performed in 3 areas: (1) mission analysis or an investigation of future earth resources applications objectives, (2) Exploratory Technology Forecasting or the identification of technology solutions looking for problems, and (3) identification of the kinds of system trade-offs that will be performed in the future in various subsystem areas. A discussion of how these three areas led to the system concepts will be presented at a later time.

Once the system concepts have been obtained, identification of their "key" technology requirements and the normative forecasting of the state-of-the-art in each of these technology areas then follows. Normative forecasting involves forecasting a technology to meet a need. (Necessity is the mother of invention.)

A separate portion of the study is constructing a methodology to assist in prioritizing the resultant technology "gaps" depending on their contributions to the various mission objectives.

At the present time, work is proceeding in the identification of technology requirements, and the final results of the study are expected to be available in June, 1978.

PLACE STUDY METHODOLOGY

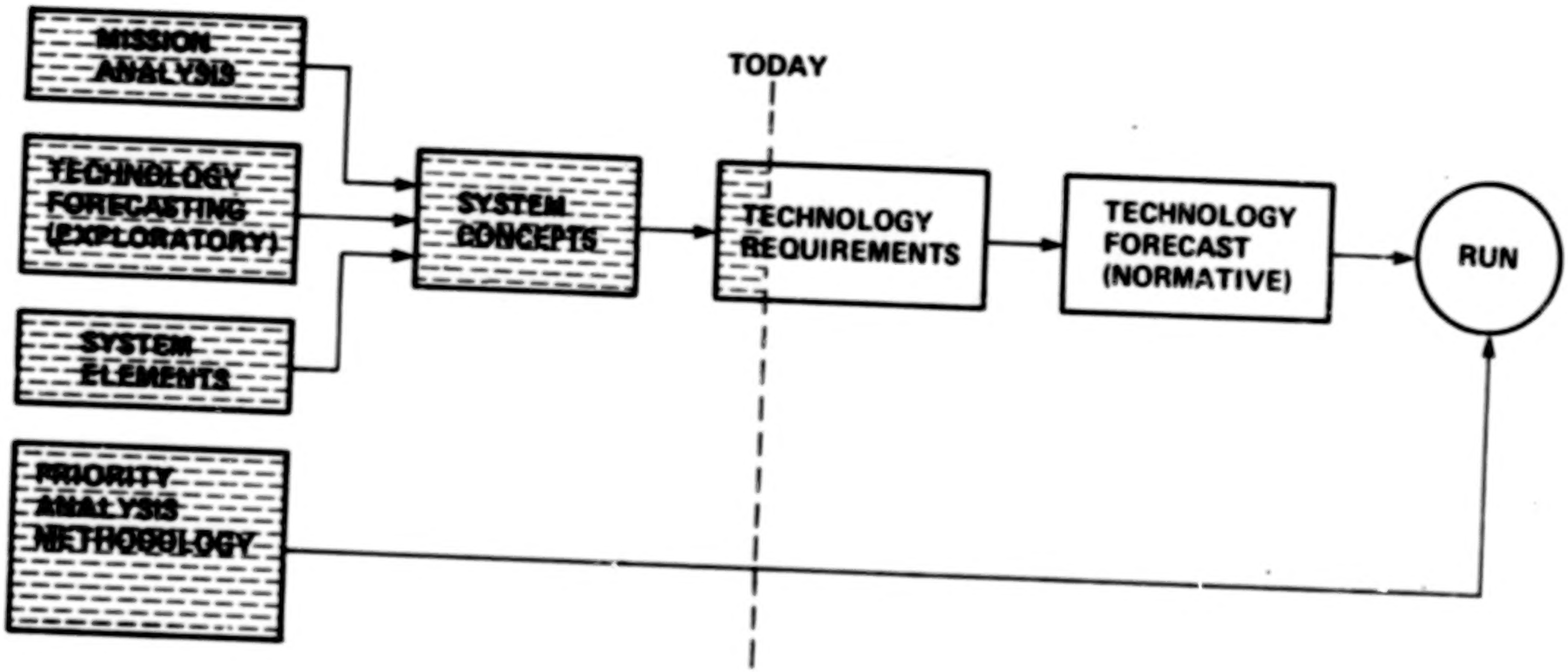

Figure 2

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(Figure 3)

This figure indicates those mission categories that were included in the PLACE study and those that were excluded. In some cases, the tie between included and excluded missions was a strong one, but the split was necessary due to limited study resources. In all cases, data from the excluded categories were assumed to be available as needed.

PLACE MISSION CATEGORIES

INCLUDED	EXCLUDED
AGRICULTURE	WEATHER
RANGE MANAGEMENT	CLIMATE
FORESTRY	ATMOSPHERIC SENSING (EXCEPT CALIBRATION)
GEOLOGICAL RESOURCES	EARTH AND OCEAN DYNAMICS
LAND USE	ENERGY/COMM/NAV
WATER RESOURCES	MILITARY
ENVIRONMENTAL QUALITY	AIRCRAFT/D.C.P.'S
DISASTER ASSESSMENT	EXTRATERRESTRIAL
	CRIMINAL ACTIVITIES (EXCEPT POLLUTION)

Figure 3

539

540

(Figure 4)

The initial list of PLACE mission objectives was compiled from a literature search of previous study results. This list was then updated by a panel of discipline scientists to include further potential future system requirements.

In order to focus the study and provide a manageable list of goals, a key set of mission objectives was selected. The criteria for selection was (1) an objective's economic and other societal importance and (2) the diversity of the specific objective and the projected diversity of its resultant technology requirements. This second criterion was employed to insure the inclusion of as wide a range of required technologies as possible.

The key set of mission objectives was approved by NASA for the purposes of this study only.

DERIVATION OF MISSION OBJECTIVES

- GE FORMULATION OF KEY MISSION OBJECTIVES
  - LITERATURE SEARCH
  - PREVIOUS STUDY RESULTS
  - MISSION REQUIREMENTS BOARD
- SELECTION CRITERIA
  - ECONOMIC AND OTHER SOCIETAL IMPORTANCE
  - DIVERSITY: OBJECTIVES, TECHNOLOGY REQUIREMENTS
- OBJECTIVES APPROVED BY NASA FOR PURPOSE OF THIS STUDY

Figure 4

(Figure 5; Figure 6)

In order to focus the study and provide a concise statement of our starting point, a set of key objectives was selected. Presented in these two viewgraphs are the 8 mission categories that cover earth resources (as defined for this study) - agriculture, range management, etc.; 8 key set objectives - crop production forecasting, grazing potential determination, etc. - and a list of mission sub-objectives that detail or explain the key-set objectives.

It should be noted that these objectives were approved by NASA for use in this study only, and do not represent official opinion for use in other endeavors.

KEY OBJECTIVES

- **AGRICULTURE — CROP PRODUCTION FORECASTING**
  - IDENTIFY CROPS
  - MEASURE ACREAGE
  - ESTIMATE YIELD
  - MEASURE PRODUCTION
- **RANGE MANAGEMENT — GRAZING POTENTIAL DETERMINATION**
  - EVALUATE STATUS AND MEASURE CARRYING CAPACITY
  - ESTIMATE LIVESTOCK COUNT
  - ESTIMATE FORAGE PALATABILITY
- **FORESTRY — TIMBER STAND VOLUME ESTIMATION**
  - IDENTIFY TREES
  - EVALUATE QUANTITY AND QUALITY OF TIMBER
- **GEOLOGY — GEOLOGICAL RESOURCES LOCATION**
  - LOCATE ORES
  - LOCATE FOSSIL FUELS
  - LOCATE CONSTRUCTION MATERIALS
  - LOCATE GEOTHERMAL RESOURCES

Figure 5

544

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KEY OBJECTIVES

- **LAND USE — LAND USE AND CENSUS ENUMERATION**
  - CREATE THEMATIC AND LAND USE MAPS
  - DETECT CHANGE IN LAND USE
  - ESTIMATE POPULATION
- **WATER RESOURCES — WATERSHED MONITORING**
  - MONITOR SURFACE SUPPLY OF FRESH WATER
  - MEASURE GROUNDWATER FLOW AND STORAGE
  - INTEGRATE RAINFALL AND EVAPORATION DATA
- **ENVIRONMENTAL QUALITY — WATER POLLUTION DETECTION**
  - DETECT, MONITOR, AND TRACE FRESH WATER POLLUTANTS
  - MONITOR EUTROPHICATION
  - MEASURE SALT WATER INCURSION
- **DISASTER ASSESSMENT — ABRUPT EVENT EVALUATION**
  - MONITOR AND ASSESS DISASTERS
  - MONITOR NON-CALAMITOUS ABRUPT EVENTS

Figure 6

545

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(Figure 7)

We have been asked by NASA to be imaginative in our forecasting of what is possible in the future, to investigate what "can be" rather than what "will be". Presented in this figure is a "credibility continuum"; it will be used to illustrate the area in which we have been asked to work - an area we call the semi-credible.

The first thing to note is that the border between the semi-credible (or what may happen) and the incredible (or what could not happen) is a subjective one. For different observers, this line will move to the right or to the left.

Presented on the chart are the authors' view of some objectives, systems, and technology areas and our opinion of their positions on this continuum. Since we are exercising in prophesy in this study, we must state that the system concepts and technology developments proposed are the opinion of the authors and have not been endorsed by NASA or by the General Electric Company.

NASA MANDATE FOR VISION

- **STUDY SHOULD ATTEMPT TO EMPLOY IMAGINATION, VISION AND INSPIRATION. SEEK TO GO BEYOND THE CREDIBLE TO THE 'SEMI-CREDIBLE'**
- **IN BOTH MISSION ANALYSIS AND TECHNOLOGY FORECASTING — ASK WHAT 'CAN BE' RATHER THAN WHAT 'WILL BE.'**

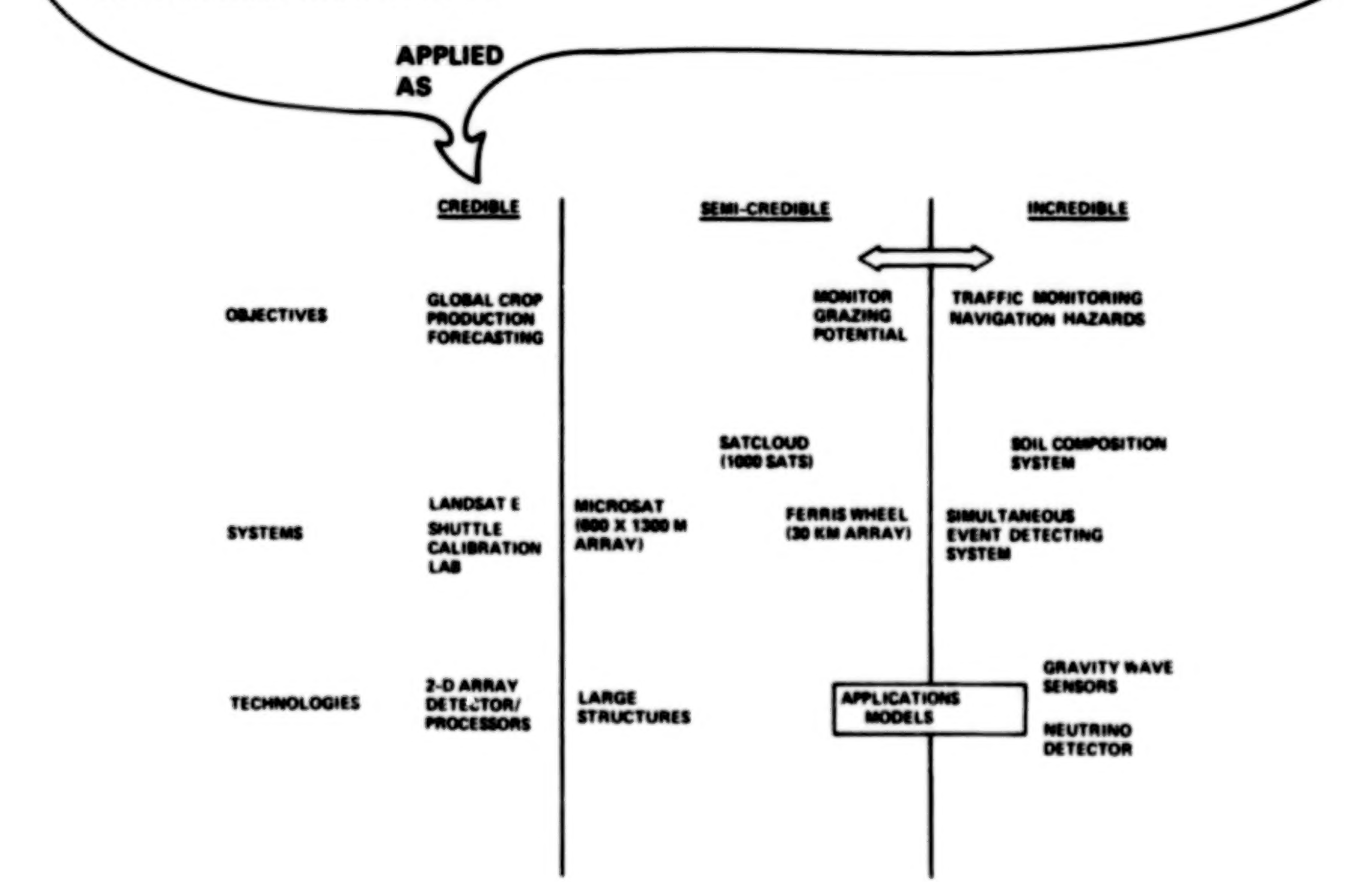

Figure 7

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(Figure 8)

A series of activities in the area of exploratory technology forecasting was performed in order to help project the study activities into the future time period.

EXPLORATORY TECHNOLOGY FORECASTING

- **OBJECTIVE**
  - IDENTIFY MISSION ENABLING CONCEPTS
- **METHODS**
  - "BLUE SKY" MEETINGS
  - LITERATURE SEARCH
  - PERSONAL CONTACTS
  - "IMAGINEERING"
- **RESULTS**
  - SENSING CONCEPTS
  - PLATFORM/SUPPORT CONCEPTS
  - DATA SYSTEM CONCEPTS

Figure 8

549

(Figure 9)

This figure indicates the method in which the system concepts were arrived at. The top line indicates a typical system engineering design procedure. However, the required links for this procedure are not known. That is, given an applications objective today, the user requirements are not precisely known, (although there are many opinions of these). Similarly, if the user requirements were known, then the optimum system specifications are still not known. That is, the relationship between phenomena and observables is not well enough understood.

An alternate procedure (3rd line) would be to forecast what technology will be available in this time frame and put together the best possible systems independent of application. However, this would be begging the question of the study since our prime objective is to identify key technology areas.

The procedure employed in the PLACE study was to employ both of these methodologies and combine needs and solution trends with engineering and scientific judgment to arrive at the system concepts to be employed.

FORMATION OF SYSTEM CONCEPTS

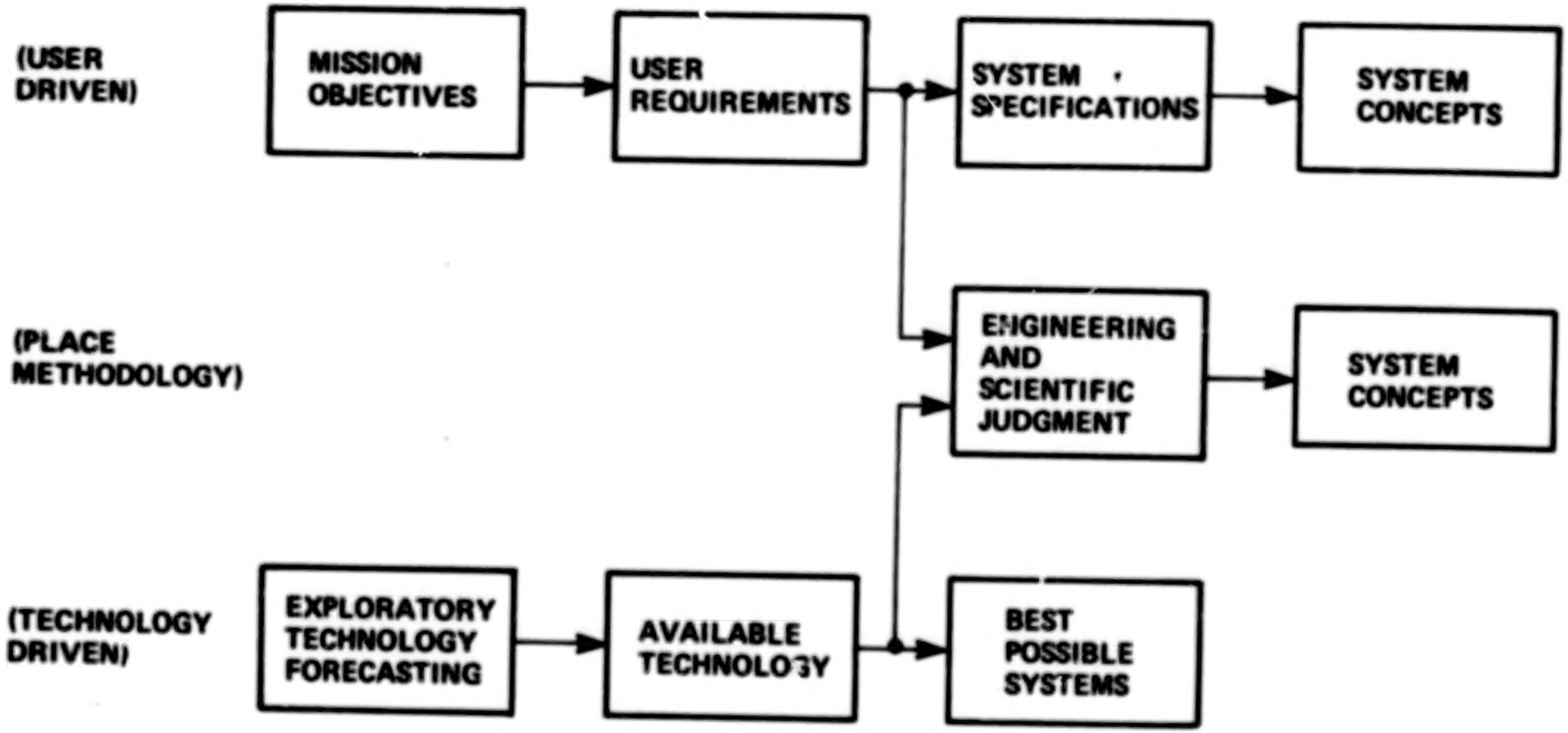

Figure 9

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(Figure 10)

The system concepts identified will be end-to-end systems, from sensor through the user. The large space structure system concepts will be identified both as part of the sensor element and the platform element.

SYSTEM ELEMENTS

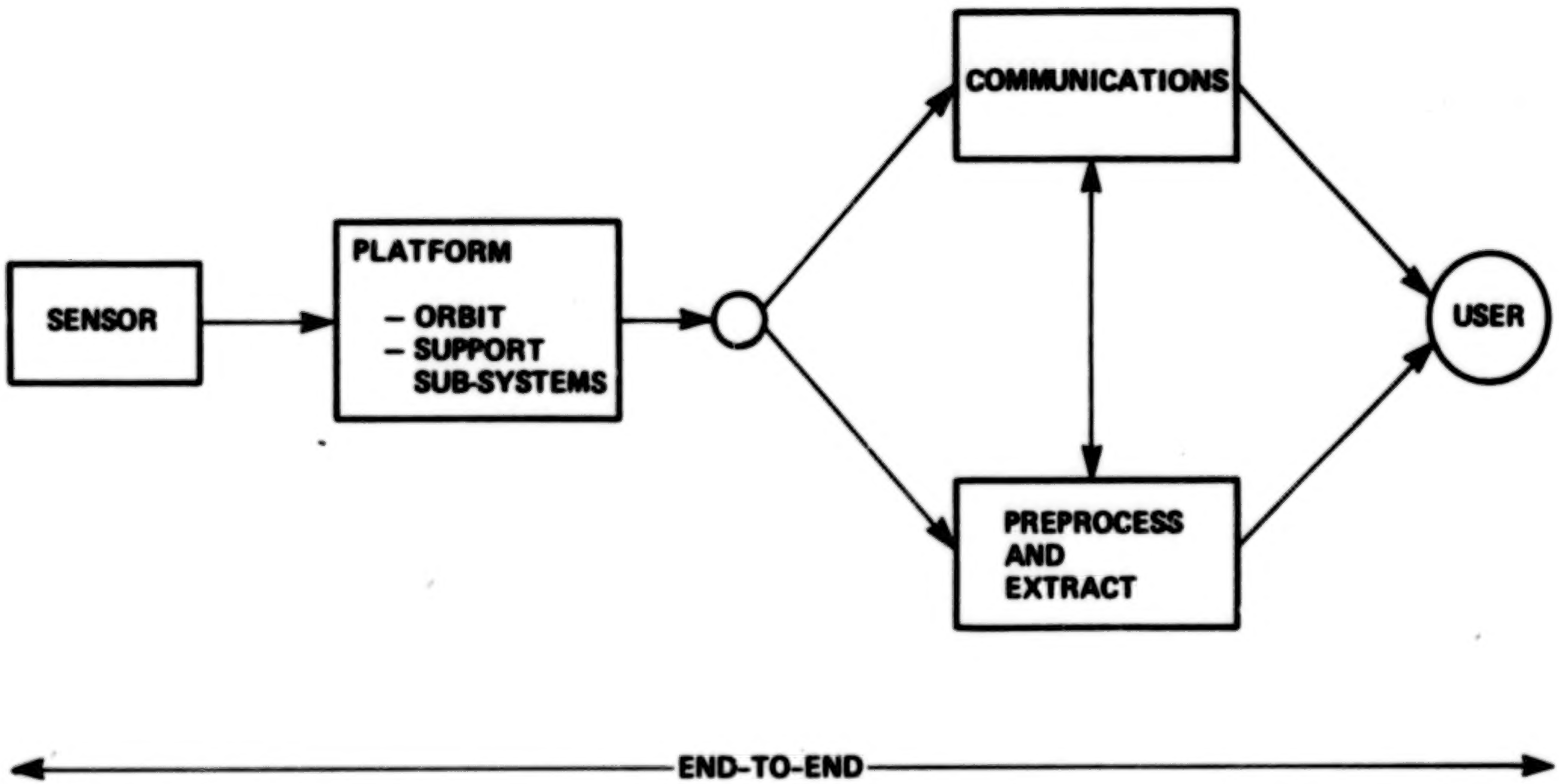

Figure 10

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(Figure 11)

The accompanying figure shows a list of sensor and system concepts being considered by the PLACE study. Each of these system concepts poses a variety of challenging technological requirements in many areas such as data processing, optics, advanced sensors, etc. The three systems with asterisks encompass a range of requirements for large space structures, and will be discussed in more detail.

SENSOR AND SYSTEM CONCEPTS

1. LANDSAT-H
2. EARTHWATCH
3. SEOS
*4. TEXTUROMETER
5. HCMM FOLLOW-ON
6. NITE-LITE
*7. MICROSAT
8. PARASOL RADIOMETER
9. RADAR ELLIPSOMETER
*10. FERRIS WHEEL RADAR
11. SATCLOUD
12. RADAR ALTIMETER
13. SWEEP FREQUENCY RADAR
14. GEOSYNCHRONOUS SAR
15. RADAR HOLOGRAPHER
16. FARADAY MAGSAT
17. TETHERSAT
18. SHUTTLE CALIBRATION FACILITY
19. OPERATIONAL SHUTTLE FLIGHTS

Figure 11

555

(Figure 12)

The first system, called MICROSAT, is based on some previous work done by GE for Langley's LSST program. It consists of a large (600 x 1300 meters) L-Band passive radiometer with simultaneous beams formed by feed horns in a focal arc, as shown.

MICROSAT – SYSTEM CONCEPT

- L-BAND PASSIVE RADIOMETER
- PARABOLIC TORUS ANTENNA WITH CLUSTER OF FEED HORNS IN A FOCAL ARC
- WOULD REQUIRE PREVIOUS COMMITMENT TOWARD LARGE STRUCTURES IN SPACE

Figure 12

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558

(Figure 13)

Some of the performance parameters for the MICROSAT system are as indicated. Its prime applications objective is the measurement of soil moisture. An approximate time projection for the system is 1990, and its measure of risk, defined as its position on the semi-credible continuum, is medium.

MICROSAT — SYSTEM CONCEPT (CONTINUED)

PERFORMANCE PARAMETERS

- FREQUENCY IS 1.4 GHZ (L BAND)
- ANTENNA SIZE APPROXIMATELY 600M X 1300M
- GROUND RESOLUTION — 1KM, ORBIT — 1000KM, REPEAT CYCLE — 3 DAYS (2 SPACECRAFT), RADIOMETRIC TEMP. RES. — $1^{\circ}K$
- DATA RATE (PEAK) — 59 KBPS

SYSTEM CONSIDERATIONS

OBJECTIVE CONTRIBUTED TO:
CROP PRODUCTION FORECASTING
GRAZING POTENTIAL DETERMINATION
WATERSHED MONITORING

RELATED SPACECRAFT:
ALL MICROWAVE SYSTEMS
EXCEPT TEXTURE SYSTEMS

TIME PROJECTION: 1988-1992

MEASURE OF RISK: MEDIUM

Figure 13

559

560

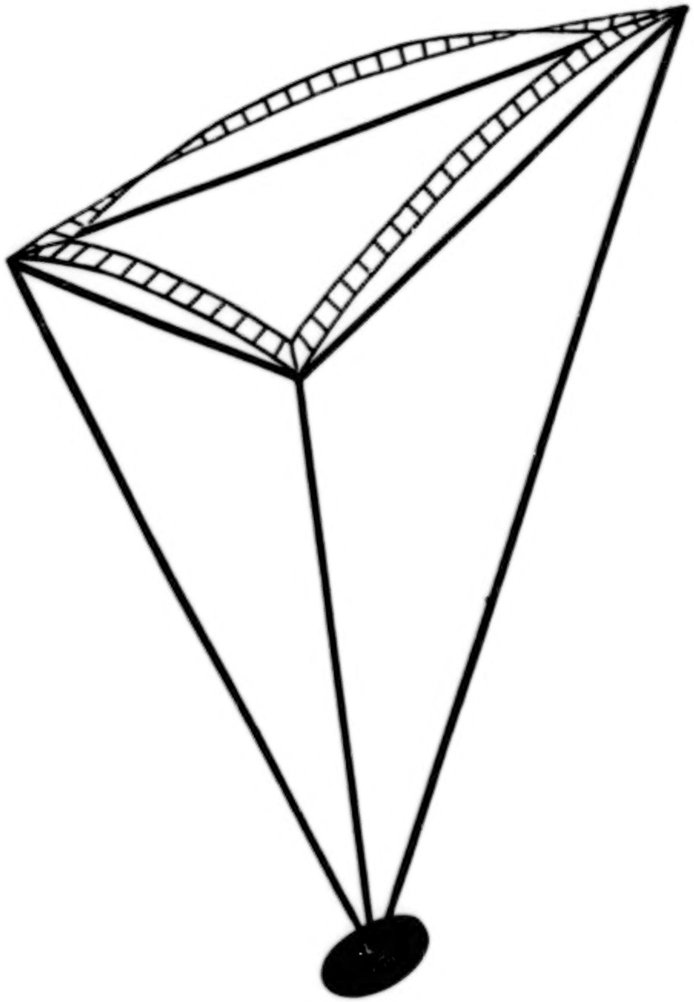
(Figure 14)

The Texturometer is a future earth resources system designed to measure ground texture in two different ways. Though there isn't much interest at the present time in using texture as a classification aid, since the human eye makes significant use of this type of information, it would seem appropriate for future space systems also.

The first method of sensing a measure of ground texture involves the range gating of a laser; since there is no requirement here for a large structure, we will not go into detail.

The second method requires the three linear banks of 100 mirrors, each mirror about 3 m^2, in order to make point measurements of texture from a few millimeters to a meter. The dimension from the mirror center to the focal plane is about 600 m. The output of the system, after a sophisticated amount of processing, will be a kind of spatial signature, analogous to the more commonly used spectral signature.

TEXTUROMETER – SYSTEM CONCEPT



- MEASURES THE TEXTURE OF THE GROUND SURFACE AT SCALES FROM 1 MM TO 1M
- SPATIAL FREQUENCY WOULD ASSIST IN CLASSIFICATION OF GROUND MATERIALS - METHOD CURRENTLY NOT PURSUED

METHOD 1

- VISIBLE/IR LASER USED AS A SCATTEROMETER – PULSES RANGE GATED TO ACHIEVE SPATIAL FREQUENCIES
- STATISTICAL MEASURE OF GROUND PERIODICITY IS THE DESIRED OUTPUT

METHOD 2

- VARIATION IN REFLECTANCE IN THREE DIRECTIONS (60° APART) PROVIDES A POINT SAMPLE OF TEXTURE
- THREE LINES OF MIRRORS, EACH CONTAINING ADAPTIVE OPTICS, PROVIDE THE MEASUREMENTS
- COMPLEX DATA PROCESSING IS REQUIRED TO TRANSFORM THE DATA TO SPATIAL FREQUENCY DISTRIBUTION
- ATMOSPHERIC SCATTERING MAY LIMIT RESOLUTION

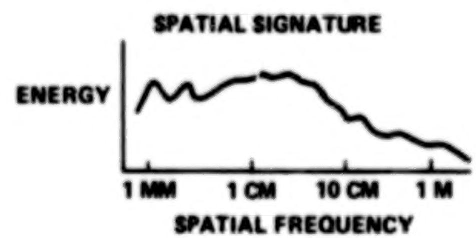


Figure 14

561

562

(Figure 15)

Some of the system performance parameters are as indicated. It is difficult to assess the value of these texture measurements to the various applications disciplines at the present, since so little work has been done in this area in the past. However, it is the opinion of the authors that this information would prove exceedingly useful in a variety of application areas.

Still a third method of measuring indicators of ground texture involves the Sweep Frequency Radar, a separate PLACE system, that will not be discussed.

The projection of system implementation is 1995-2000, and its measure of conceptual risk (semi-credibility) is quite high.

TEXTUROMETER — SYSTEM CONCEPT (CONTINUED)

PERFORMANCE PARAMETERS

- USE EITHER CO_2 (9-11 μm) OR Nd/YAG (1.064 μm)
- REQUIRES PICOSECOND PULSES
- DATA RATE — 25 SAMPLES - 90 KBPS
- EACH LINE OF MIRRORS CONTAINS 100 MIRRORS, EACH 3.0 METERS SQUARE
- ORBIT IS 600 KM CIRCULAR
- OPTICAL SPECTRUM: VISIBLE THROUGH IR
- ADAPTIVE OPTICS AND IMAGE MOTION COMPENSATION REQUIRED
- MIRROR FOCAL LENGTH ≈ 600 M, MIRROR LINE LENGTH 300 M

SYSTEM CONSIDERATIONS

OBJECTIVES CONTRIBUTED TO:
ALL KEY OBJECTIVES

RELATED SPACECRAFT: SWEEP FREQUENCY RADAR

TIME PROJECTION: 1995

MEASURE OF RISK: HIGH

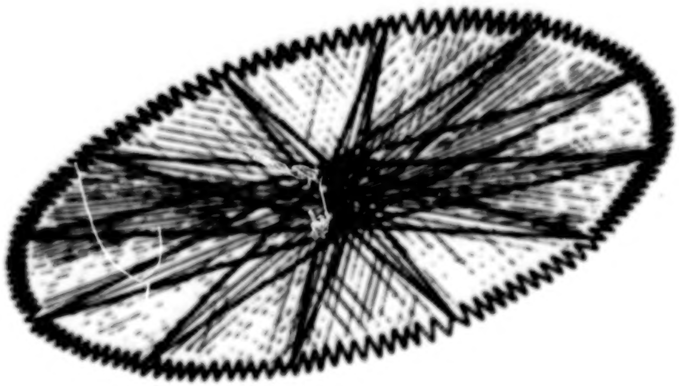
Figure 15

(Figure 16)

The third PLACE system requiring a large space structure is the Ferris Wheel Radar System. As the name may imply, this is a large rotating real aperture radar system, whose prime purpose is penetration of the ground in order to map boundary layers. Since the spacecraft spin vector is fixed in inertial space, the system is only operable over a limited range of latitude.

The structure relies on cable tension for support and poses challenging requirements in the areas of deployment and control. Preliminary mass calculations indicate that the system could be transported into LEO in two shuttle loads.

FERRIS WHEEL RADAR — SYSTEM CONCEPT



- LARGE (30 KM DIAMETER) ROTATING CABLE STRUCTURE THAT RELIES ON CABLE TENSION FOR SUPPORT. PRESUMES PREVIOUS COMMITMENT TO ASSEMBLY OF LARGE STRUCTURES IN SPACE
- REAL APERTURE RADAR OPERATES AT LOW FREQUENCY (30-300 MHZ) FOR GROUND PENETRATION
- RESULTANT RETURN SIGNAL CAN MAP SUBSURFACE FEATURES (BOUNDARY LAYERS AND GROUNDWATER)
- SPACECRAFT SPIN VECTOR IS FIXED IN INERTIAL SPACE
- PROBLEM AREA TO BE EXAMINED IS THE ATTENUATION EFFECTS OF THE IONOSPHERE.
- IC CHIPS FORM ELEMENTS OF A RANDOM SPARSE PHASED ARRAY
- TRADE BETWEEN CW AND PULSED IMPLEMENTATION YET TO BE PERFORMED

Figure 16

CONTENTS

VOLUME I

(Figure 17)

The radar system will be implemented using a sparse phased array with communication and timing provided by optical methods. Some performance parameters are listed in the accompanying table.

The prime objectives of such a system would be geologic resources location and the inference of subsurface moisture. Again, the time projection is late in the period, 1995-2000, and the measure of conceptual risk quite high.

Please note that, as presented, these are system concepts and not system designs.

FERRIS WHEEL RADAR — SYSTEM CONCEPT (CONTINUED)

PERFORMANCE PARAMETERS

- 30 KM DIAMETER
- 1M TO GREATER THAN 75M DEPTH
- 300 M GROUND RESOLUTION
- VERTICAL TARGET RESOLUTION BELOW GROUND SURFACE — APPROXIMATELY 2M
- FREQUENCY — 30-300 MHZ
- SPIN RATE APPROXIMATELY 1 REV/HR
- 900 KM ORBIT

SYSTEM CONSIDERATIONS

OBJECTIVES CONTRIBUTED TO:
GEOLOGIC RESOURCES LOCATION
WATERSHED MONITORING

COMPETING SPACECRAFT: NONE

TIME PROJECTION: 1995-2000

MEASURE OF RISK: HIGH

Figure 17

567

568

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16. Abstract
These proceedings include a compilation of the papers presented during a seminar which provided Government and Industry representatives an opportunity to exchange information, to review the present status of related technology, and to plan the development of new technology for Large Space Systems. These papers were divided generally into two major areas of interest: the first group addressed subjects pertinent to large antenna systems; the second group addressed technology related to large space platform systems.

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